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TITLE

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C-130 - RESULTS OF CENTER WING
RESIDUAL STRENGTH AND CRACK
PROPAGATION TEST PROGRAM

SUBMITTED UNDER

NAS-1-9485 LSR NO. 7221-70-012A

MODEL C-130 REFERENCE JOB NO. 74C330

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ABSTRACT

Fourteen C-130 airplane center wings which had experienced from approximately 4,000 to 13,000 hours of flight service and its associated fatigue damage were tested to destruction, seven in upbending and seven in downbending. Six wings were tested directly for static residual strength in the fatigue-damaged condition as received from field service. The other eight wings were tested in crack propagation cyclic testing at a prescribed stress level for a maximum of 10,000 cycles. Then the stress level was reduced and testing was continued up to a maximum of 20,000 total cycles. Cyclic testing was performed with constant-amplitude stresses at a stress ratio of +0.1. Maximum cyclic skin stresses were approximately 18,000 psi. At the conclusion of cyclic testing, a static test to destruction was conducted to determine the residual strength of each fatigue-damaged specimen.

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GENERAL SECTION

ABSTRACT

The General Section, Part G, contains details of test procedure, data acquisition, methods of loading and control, test specimen configurations, applied loads, Table footnotes for subsequent sections, and other details which are common to the fourteen specimens.

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REFERENCES

1. Laboratory Service Request Number 7221-70-012A, "C-130 Center Wing Residual Strength and Crack Propagation Test", S. K. Brown, Lockheed-Georgia Company, February 16, 1970.

2. Test Data - Laboratory Data Record Book Numbers: A371602-A371615, F. L. Reeder; A371751-A371762, A371551-371600, W. Dirkin, dated 10-23-70 through 6-17-71.

3. ER 9477, Vol. II, C-130 Fatigue Life Monitoring Program Methodology - Final Report, Phases II and III, T. H. Williams and E. E. Blount, Lockheed-Georgia Company, October 31, 1968.

GLOSSARY

BEAM RUNOUT - The end area of a reinforcing beam used to stiffen the lower surface access door cutout. The term can be used to indicate either the inboard end (viz. W.S. 58) or the outboard end (viz. W.S. 176-182). There are two such beams forward of each cutout and also two beams aft of each cutout.

DOUBLER RUNOUT - The end area of a reinforcing doubler used to stiffen a cutout. The lower surface has a doubler for each of the two access door cutouts. The upper surface has a doubler for each of the two outboard access door cutouts, each of the two fuel filler cutouts, and the center access door cutout.

INITIAL FATIGUE DAMAGE - Refers to any in-service fatigue damage found to exist in the test specimen after it was received.

WING JOINT BOLT NODE - One of the 13 upper surface or 15 lower surface center wing to outer wing main attachment bolt locations.

SLINGSHOT DOUBLER - External skin doubler used to repair fatigue cracks in skin - so-called because of their appearance.

FAILURE LINE - Path of final failure of a specimen that occurred during the residual strength test.

MINOR CRACK - Any crack of approximately one inch length or less.

RAINBOW FITTING - W.S. 220 upper or lower joint fitting which mates with the center wing.

NOMINAL STRESS - Stress calculated with the formula, $s = \frac{M c}{I}$.

INTRODUCTION

This program was conducted to determine the residual strength and crack propagation characteristics of 14 C-130 center wings containing service imposed fatigue damage. Eight of the wings were subjected to crack propagation tests before conducting the residual strength test. Only the surface which was loaded in tension was considered as the test specimen. Failure always occurred in that surface. Seven of the specimens were tested with upbending loads producing tension in the lower surface and seven with downbending loads producing tension in the upper surface.

This test program, authorized by Reference 1, was conducted in the General Structural and Materials Laboratory Department of the Lockheed-Georgia Company, Marietta, Georgia, from October 23, 1970 through June 17, 1971. Laboratory Data Record books, Reference 2, were used to record test data.

Selected scenes from the wing box test program are shown in motion picture film with color and sound narration. This is available through the Motion Picture and Television Department.

OBJECT

The objective of this program was to determine static and fatigue crack propagation characteristics in a major structural component, a center wing box, containing service induced fatigue damage.

SUMMARY OF RESULTS

Static residual strength of the specimens as received ranged from 98% to 117% limit load. Several of these specimens had initial crack lengths of 4.0" or more. The static residual strength of the wings which had been subjected to up to 20,000 crack propagation test cycles, and were usually severely damaged, ranged from 56% to 87% limit load.

TEST SPECIMEN

Description

The structural configurations of the C-130 center wing boxes used as test specimens for this program are illustrated in Appendix A. Gross dimensions of the center wing box test specimens are 440 inches length, 80 inches chord, and 32 inches depth. Each specimen weighs approximately 3800 lbs.

The upper surface of the wing box is composed of four panels. Each panel is 440 inches in span and 20 inches in chord, and is fabricated from machined 7178-T6 aluminum extrusions which have six integral risers spaced at 3.3 inch intervals. Each of these panels is further stiffened by the installation of three spanwise stringers made from 7178-T6 extruded hat sections spaced at 6.6 inch intervals and installed with riveted attachments except at the spanwise splices. The spanwise splices are configured as butt joints with an extended leg of a hat section stiffener forming a splice plate and fastened with steel lockbolts. Basic upper surface skin thickness of the machined panel is 0.100 inch for the entire span of the center wing. The principal structural discontinuities are at the transition structure immediately inboard of the W.S. 220 production joint; at the W.S. 180 access door cutout corners, the W.S. 120.5 fuel filler cap opening, and at the inboard (W.S. 105) and outboard (W.S. 135.5) ends of its reinforcing doubler; at the W.S. 61.5 wing-fuselage support rib; and at the dry bay access door cutout at W.S. 1.5 and the ends of its reinforcing doubler at W.S. 34.5/37.5. These are the locations where most of the upper surface fatigue cracks originated.

The lower surface is composed of three panels. Each panel is 440 inches in span and 26.7 inches in chord, and is fabricated from chem-milled 7075-T6 plate with extruded 7075-T6 hat section stiffeners located at 5.70 inch spacing. The spanwise splices and attachments for the lower surface are similar to those for the upper surface. Lower surface skin thickness of the machined panel is 0.155 inches in the center region between W.S. 68L and W.S. 68R, tapers from 0.155 inches at W.S. 68 (both sides) to 0.092 inches at W.S. 179, and remains at 0.092 inches to W.S. 220. The principal structural discontinuities, which occur approximately symmetrically in both sides, are in the transition structure immediately inboard of the W.S. 220 production joint; at the W.S. 120.5 fuel bag access door; and at the ends of the reinforcing I-beam (W.S. 181/176 and W.S. 58) and doubler (W.S. 168) which extend past both sides of the W.S. 120.5 fuel bag access door.

The front and rear spars are composed of 7075T-6 aluminum extruded caps with 7075-T6 webs; except in the area of the nacelle where the webs are 301 Full Hard, 17-7PH, or AM 350 stainless steel (dependent on specific aircraft serialization).

Flight Service Experience

The flight hours of service utilization accumulated on each airplane from which the center wing box test specimens were taken are shown in Table I. The twelve military aircraft were flown on nine types of missions. The number of flight hours for each type of mission; a

definition of each mission in terms of flight time versus altitude, airspeed, fuel, and cargo; and other related flight service experience details are given in Appendix B. Utilization of the commercial specimens was similar to two of the military missions.

Preparation

Center wing boxes were tentatively selected for testing based on the Air Force field inspection reports of fatigue cracking, and followed by visual inspection on the C-130 Modification production line to verify or further define the extent of fatigue cracking experienced in service.

On some wings, substantial repair patches were found. These were removed to the extent necessary to obtain meaningful crack propagation data, and the wing restored as nearly as possible to its earlier condition of unrepaired damage.

In the tests of earlier specimens, most of which were for static residual strength tests, the elastomeric surface coating was stripped only in local regions believed to have fatigue cracks. For Specimens #7 and #10, #11, #12, #13, and #14, the entire tension surface of the wing was stripped to permit unrestricted observation of crack propagation during testing. Cracks as small as 0.1 inch length could be detected without difficulty by visual inspection of critical regions of the stripped surface.

TEST ARRANGEMENTS AND PROCEDURE

Test Fixture

The specimens were tested in a suitable test fixture which was designed and constructed at the Lockheed-Georgia Company at Marietta, Georgia. Test loads were applied through this fixture, which is illustrated in Figure 1. Shear and bending moments were applied at both ends of the specimen (W.S. 220R and 220L) and reacted by balancing forces at W.S. 61R and 61L. Torsion was generated by applying the resultant shear forces at prescribed chordwise locations at W.S. 220R and 220L. These loads were applied to the wing through the test fixture by means of hydraulic jacks attached to the fixture, which were in turn controlled with a servo-control mechanism. The servo-control system consisted of a dual bridge load cell, a "Servac" or load control module, and a servo valve, or hydro-electrical proportional pressure valve. The Servac receives an electronic signal from the load cell, which it compares with an adjustable command signal, and adjusts the electronic signal to the servo valve so as to maintain the required pressure in the hydraulic

jack, and hence the force on the load cell, which is transmitted into the fixture. The second readout bridge on the dual bridge load cell is used as an independent check on the load being transmitted to the fixture. Increasing load is accomplished by increasing the command signal, varying load by varying the command signal with a sine wave generator. The load cells are physically connected in series with the applied load and their calibrations are directly traceable to the National Bureau of Standards.

Figure 1 is a sketch showing the general configuration of the text fixture. Figures 2 through 4 are photographs showing the empty test fixture, the test fixture with a specimen installed, and the test fixture with the safety screen in place, as when testing is conducted.

Instrumentation

Electrical resistance type strain gages were installed at various locations on each of the test specimens to obtain strain measurements during the tests. Appendix C to this section contains the strain gage locations and strain readings which were recorded. The strain readings were measured with either a strain indicator or a thirty-channel digital strain indicator, plus an eight-channel Offner recorder. A list of all electrical equipment utilized during the program is given in Table I.

Birefringent plastic coating was bonded to the lower surface skin in several areas of specimen number 1. Details of the birefringent stress survey are also presented in Appendix C to this section.

Loads and Testing

The load conditions applied during the crack propagation tests are shown in Table II and Figures 5 and 6, and are nearly identical in shape to the load conditions that were applied to the full-scale C-130E wing during a previously performed fatigue test program that utilized a test spectrum based on Tactical Air Command usage as shown in Figures 7 and 8. The upbending condition represents the principal loadings which induce lower surface fatigue damage. Since its shape is also nearly identical to the moment diagram for upbending static ultimate strength, this one condition, shown in Figure 5, was used for all upbending testing on this program (both cyclic crack propagation and static residual strength). For downbending, the most representative fatigue loading condition is labeled D-1, which has a significantly different loading distribution from D-2 (loading condition for static ultimate strength). D-1 was used for all downbending cyclic crack propagation and for static residual strength test of Specimen No. 6; thereafter D-2

was used for all downbending static residual strength testing (Specimens #7, #8, and after prior testing cycling with D-1, Specimens #9, #10, #11, and #12). Magnitudes of all three applied loading conditions are listed in Table II.

Approximate wing skin stresses (spanwise) on the tension surface for each of the three test loading conditions are shown in Figure 9 for 150% Limit Load and for 50% Limit Load. All cyclic crack propagation testing was conducted at 50% Limit Load or less. The stresses shown are nominal values at each span station, and do not reflect some chordwise variation in stress due to small changes in wing depth or the effects of local structural discontinuities or stress concentrations.

During the crack propagation tests a selected stress level was applied for a maximum of 10,000 cycles. Then the stress level was reduced and testing was continued up to a maximum of 20,000 total cycles. Testing was performed with constant-amplitude stresses at a stress ratio of +0.1. Maximum cyclic skin stresses were approximately 18,000 psi. Crack propagation testing was conducted at a rate ranging from two to four cycles per minute, depending on hydraulic flow requirements and the time needed for the lag in loading system response to dissipate.

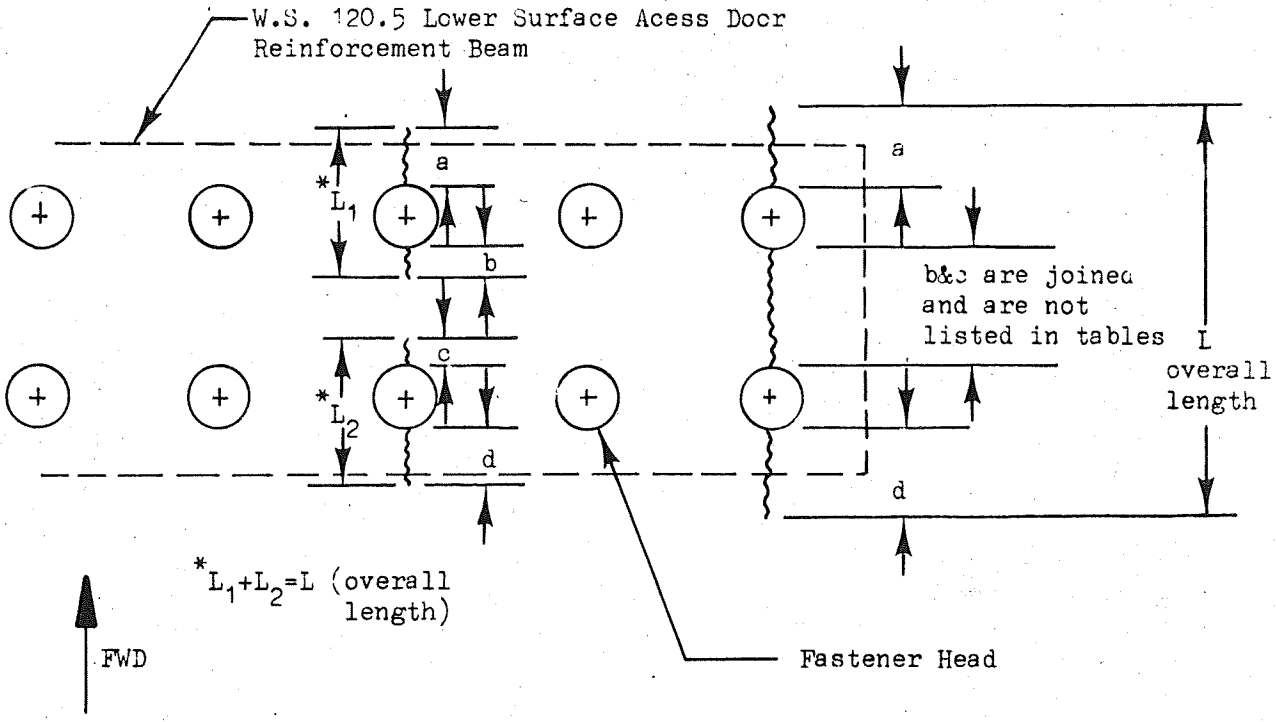
Static residual strength testing was conducted by loading the specimen in successive increments of 10% Limit Load or less until major failure across the wing cross-section tension surface occurred. Loading was immediately stopped whenever any appreciable sonic reports were heard. Load level was reduced to 20% Limit Load for safety during inspection, crack growths were measured and recorded, then testing was resumed.

Skin Cracks

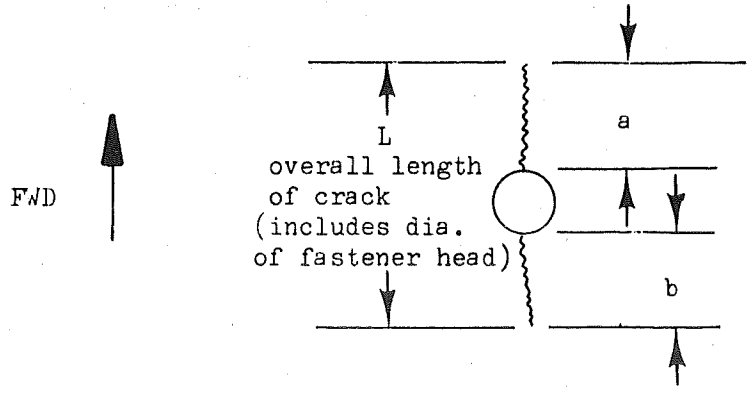
The methods used to identify and record skin cracks throughout the program are described in this section.

For convenience all skin cracks were identified with reference to associated fastener holes, using two cases: (1) Case I, the cracking common to the double row of fastener holes of the W.S. 120 lower door reinforcement beam runout, and (2) Case II, any other fastener hole on either surface. Both cases are shown schematically below and in Figures 10 and 11.

CASE I



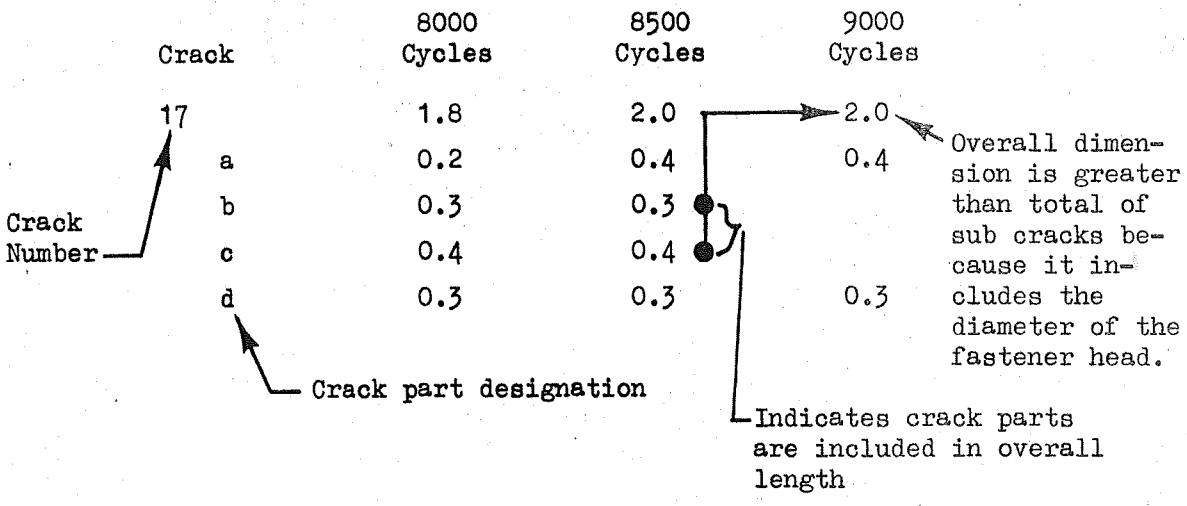
CASE II



Note: In both cases the the 'a' part of the crack is always forwardmost.

Note that in Case I the spanwise crack location varies and it is arbitrarily shown as propagating through the 1st & 3rd fastener holes from the beam end. Also, after sub items b and c join, only a, d and the overall length are subsequently recorded. In both Case I and II, a is always the forwardmost length.

Such data are presented in tabular form in this report. In recording Case I cracks, the symbols below indicate that parts b and c of the crack joined and their lengths are included in the overall length of the crack which is always recorded above the lengths of the crack parts, associated with the item number.



Example 1

Case I

Crack	7450 Cycles	8000 Cycles	8500 Cycles
21	2.0	2.4	2.6
a	0.4	0.5	0.6
b	0.2	0.4	
c	0.5	0.5	
d	0.2	0.5	0.7

Overall dimension is greater than sub crack total because it includes the diameter of the fastener head

Indicates crack parts are included in overall length

Example 2

Case I

Crack	10,000 Cycles	10,500 Cycles	10,850 Cycles
12	4.9	9.5	14.5
a	1.6	1.6	3.3
b	3.1	7.7	11.0
29	0.2	0.2	
a	-	-	
b	0.2	0.2	
27	2.7	3.2	
a	1.3	1.5	
b	1.2	1.4	

Indicates cracks 29 and 27 have been included in crack part 12b

Overall dimension is greater than sub crack total because it includes the diameter of the fastener head

Crack Number

Crack part designation

Case II

Example 1, Case I, above indicates that b joined c at 9000 cycles. In this example there was no apparent increase in overall crack length or in the lengths of b and c from 8500 to 9000 cycles. This is due to the fact that throughout the program cracks were measured only to the nearest 0.1 inch.

Example 2, Case I, indicates that b joined c at 8500 cycles. Although parts b and c still exist after joining, for clarity they are no longer listed.

In cases where the original crack had already gone through the b and c locations only a, d and the overall length were recorded throughout.

The example for Case II illustrates the method of indicating two or more cracks joining. In the example cracks 27 and 29 are shown to have joined crack 12 at 10,850 cycles. After joining, only the one crack number 12 was listed. (All examples were taken from Table I-4 of Part 4.)

In general, throughout the crack length tables in this report, crack items are arranged first according to spanwise location and secondly from forward to aft. This allows associated cracks at any given wing station, such as the failure line, to appear together in the table.

Notes for all tables of crack length data presented in the other parts of this report are itemized in Table III.

TEST RESULTS

All test results are contained in the individual parts of the report. Each part of the report is identified by the test specimen number; i.e., information for Specimen 1 is found in Part 1, etc. Tables and Figures are identified with specimen numbers also. For example, Table I-1 indicates Table I of Specimen 1.

A summary of all loads applied during the program is given in Table IV. A summary of fatigue damage in the downbending test specimens is contained in Table V, and for the upbending specimens in Table VI. A summary of failure line locations is given in Tables VII and VIII.

CONCLUSIONS

In general, crack propagation during static residual strength tests of specimens not subjected to test cycling was localized at one or two locations on each wing. For specimens which had been subjected to extensive fatigue test cycling prior to static residual strength, the extent

of cracking was much more widespread over the wing surface, and severe crack propagation tended to occur at many different locations. Locations of fatigue cracks usually occurred in the vicinity of major structural discontinuities, such as, at the termination of fastener patterns attaching reinforcing stiffeners and doublers to the skin, wing rib to surface attachments, access door cutouts, fuel filler openings, and drain holes. Initial cracks at the corners of the cutouts ranged up to three inches in length, but none of these propagated appreciably or were at locations of eventual wing static failure. The single most common feature of crack initiation which eventuated in a massive fatigue crack or static test failure was cracking which initiated at the fasteners in the terminations of reinforcing doublers and stiffeners.

The detection of cracks through any thick coating (such as the elastomeric coating used by the Air Force) proved difficult, but after the surface coating was removed cracks as small as 0.1 in. length could be detected by an experienced inspector. The greatest asset in crack detection proved to be a prior knowledge of fatigue prone locations, which suggests the importance of prompt dissemination of information on the discovery of cracks on an aircraft to all operators of that model aircraft.

Keeping in mind the small sample of specimens of any one of the several test configurations (i.e., only one specimen failed at W.S. 34.5, only two at W.S. 105/135), these conclusions are drawn:

1. Initial crack length and location in the wing configuration tested is not necessarily a true indication of future extensive fatigue crack propagation, due to design features which arrest propagation. Such features are reinforcing structure which permit redistribution of stresses in the vicinity of a crack, stopdrilled and fastener holes, and panel splices.

2. Initial surface crack length is not directly related to the residual strength the structure possesses, in the wing configuration tested. Other factors may be more pertinent in any attempt to estimate residual strength, such as presence and condition of reinforcing structure, crack tips which end in panel splices, stopdrilled holes and the like.

3. Calculations and crack propatation test results have demonstrated that the wing as received in the fatigue damaged condition from service might have sustained thousands of additional flight hours while maintaining a substantial residual strength (55% limit load or more).

4. A direct application of plate fracture mechanics is not a suitable means of predicting crack growth, due to the presence of features not accounted for in that theory. These features include reinforcing structure, uncertain stress distribution due to locally complex structural detail, the presence of crack tips in holes, and non-idealized boundary conditions.

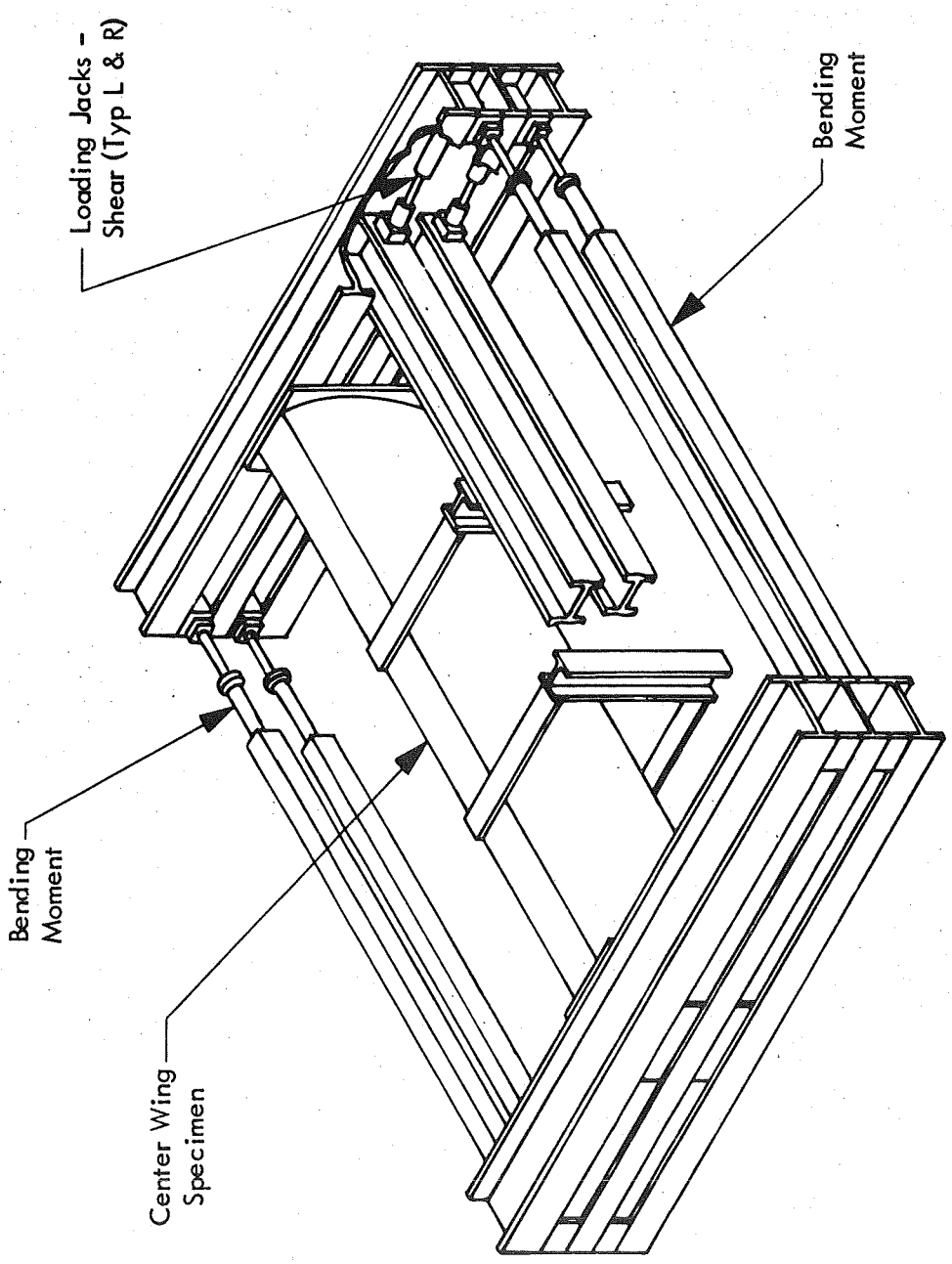


Figure 1 - Test loading fixture.

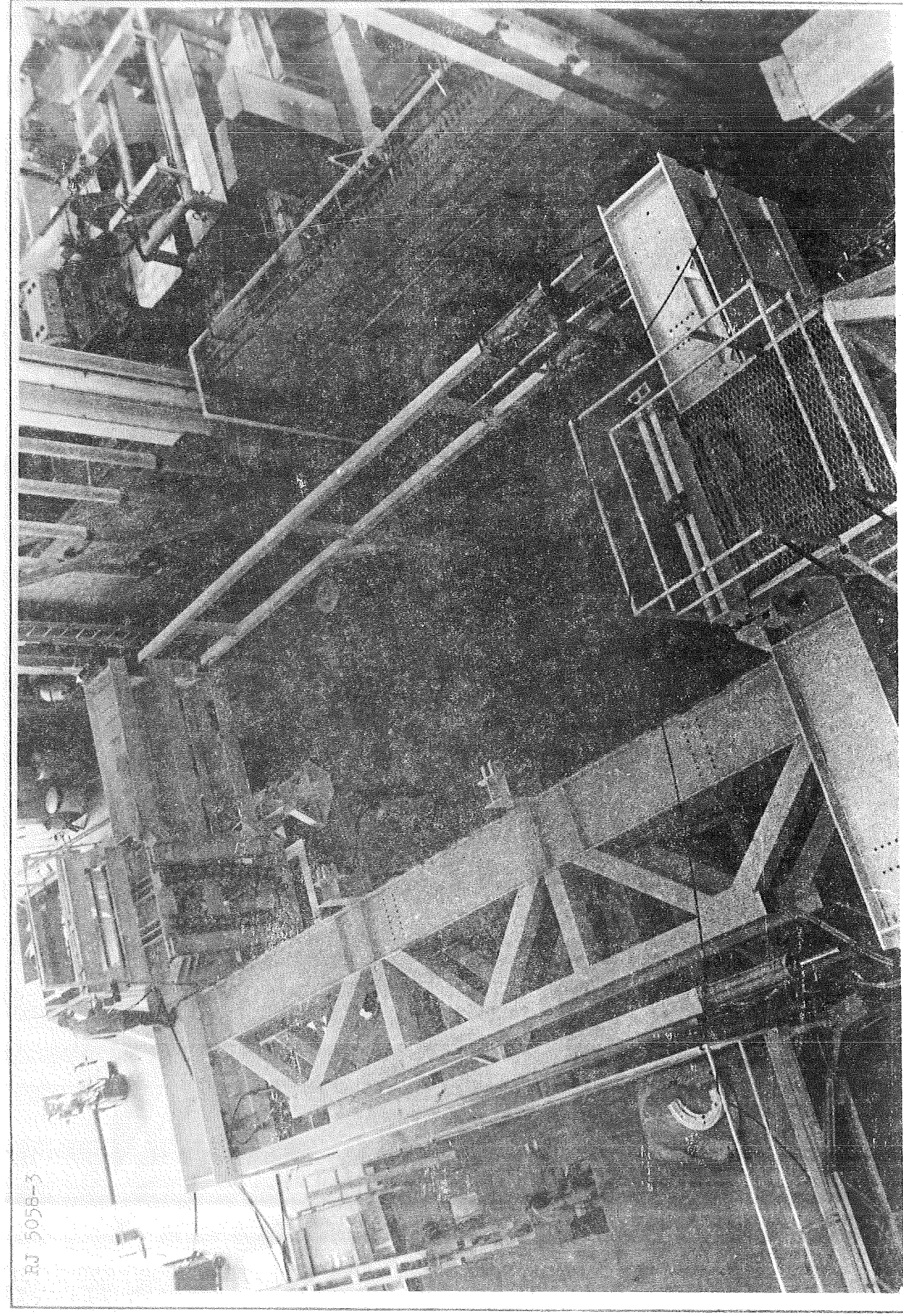


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FIGURE 2
TEST FIXTURE

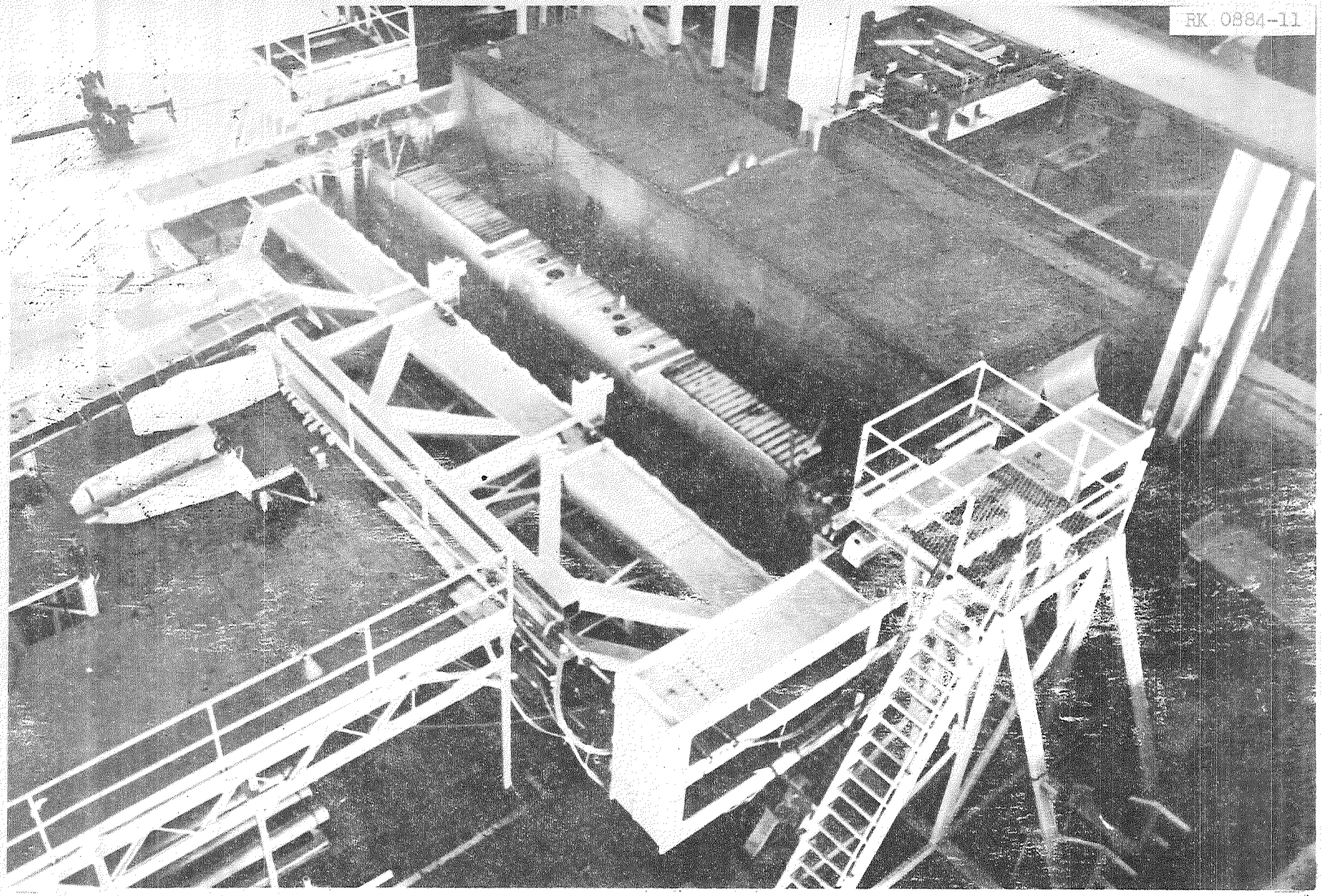


FIGURE 3

TEST SPECIMEN INSTALLED IN TEST FIXTURE

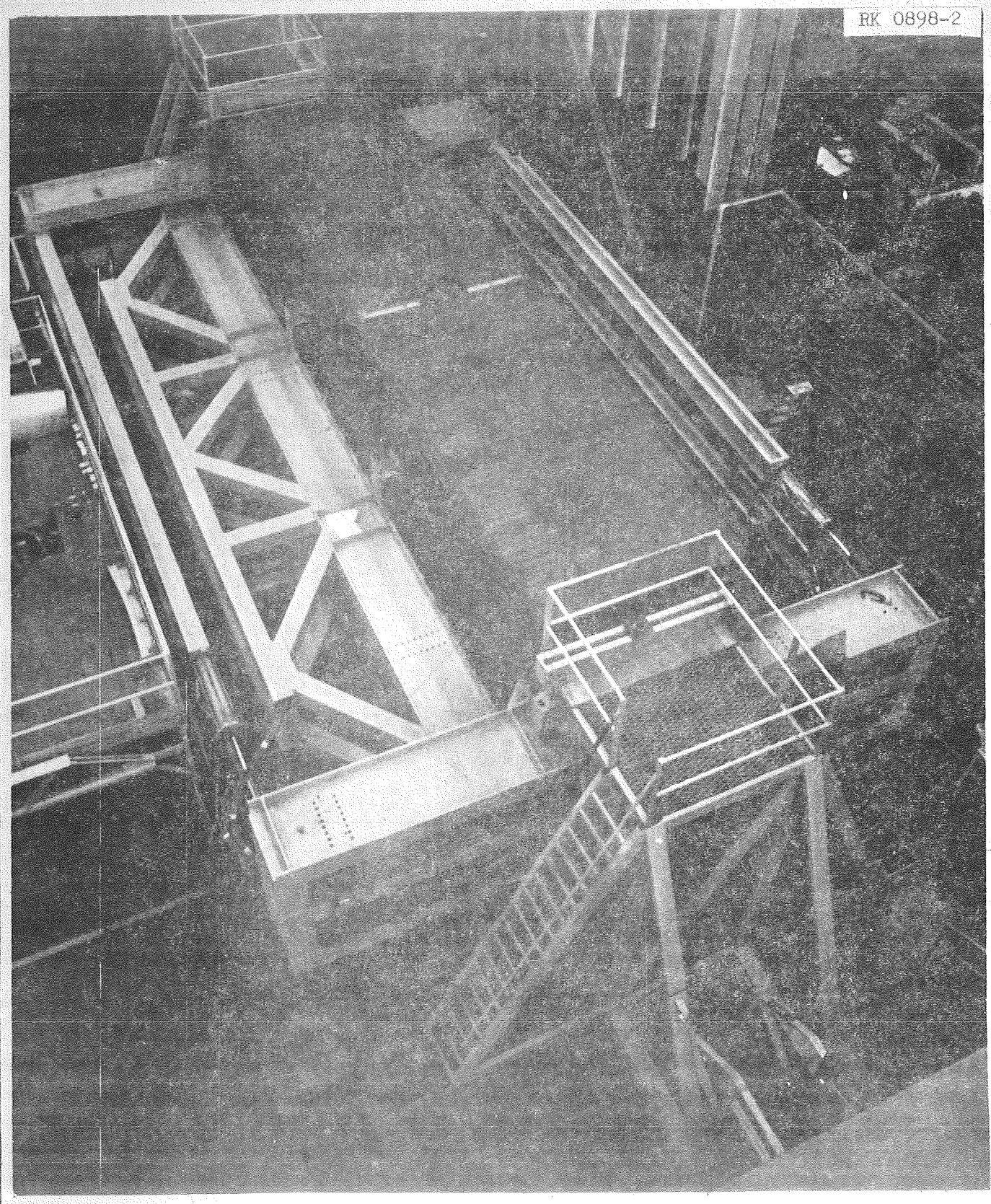


FIGURE 4

SAFETY SCREEN IN PLACE OVER TEST SPECIMEN. SETUP READY FOR TESTING.

FIGURE 5

APPLIED LOADING FOR UPBENDING TESTS
100% LIMIT LOAD

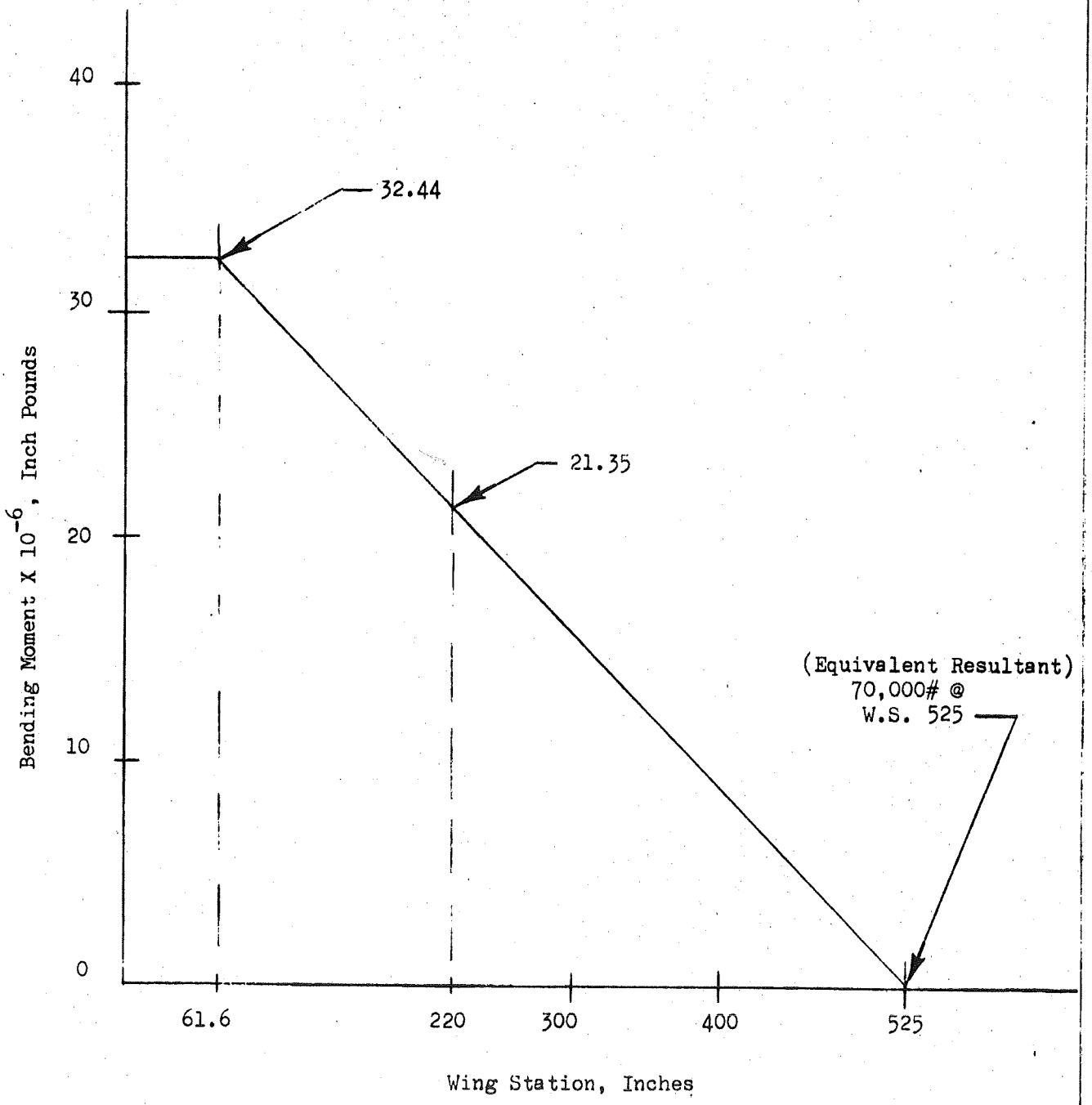
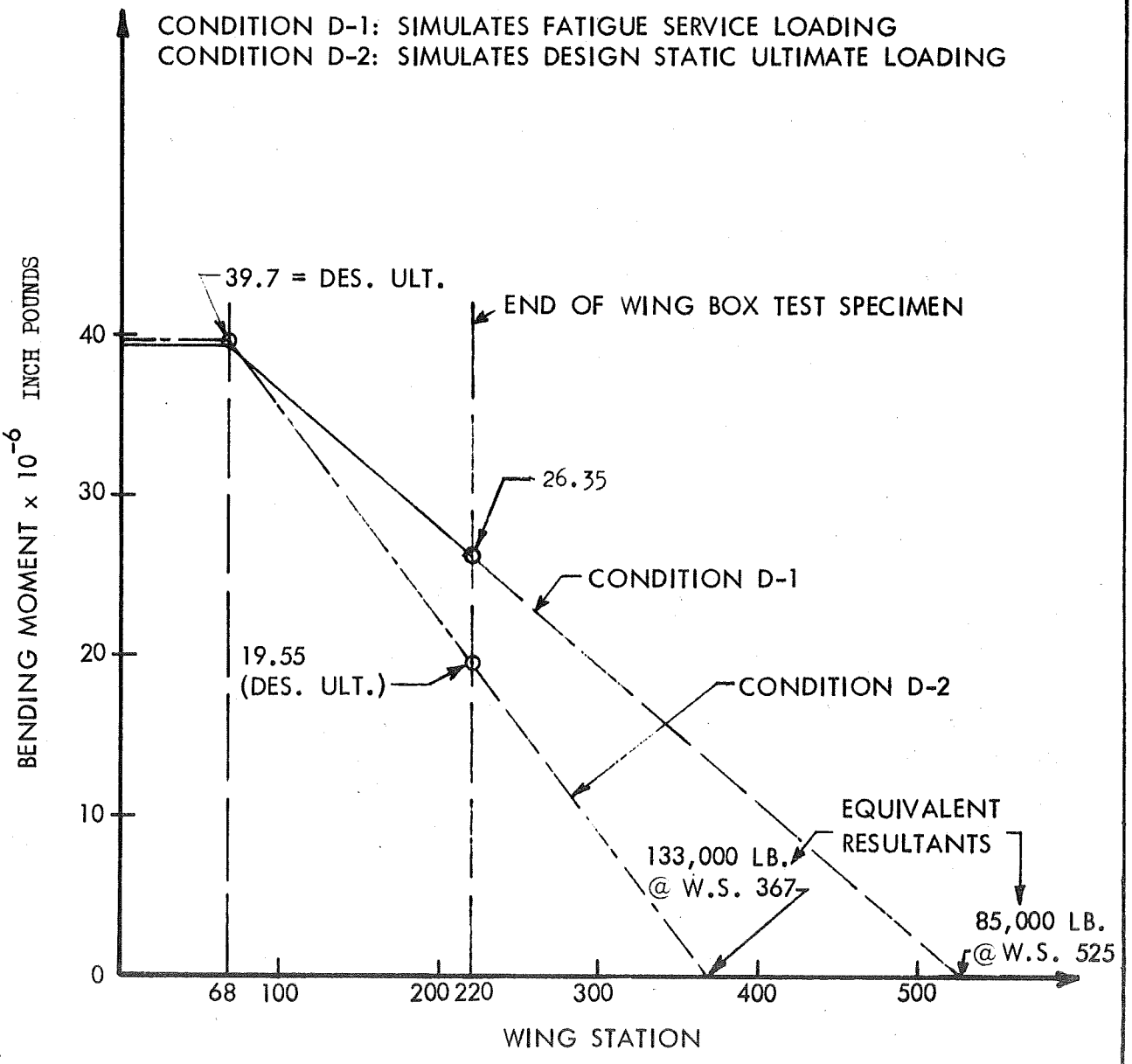


FIGURE 6

APPLIED LOADING FOR DOWNBENDING TESTS
150% LIMIT LOAD



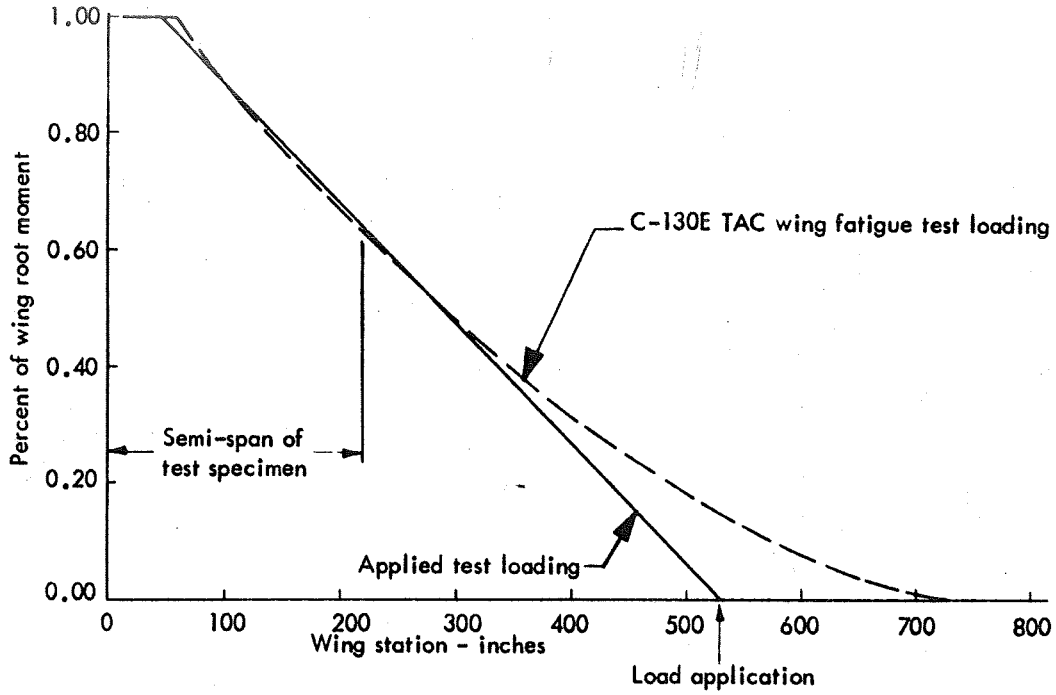


Figure 7 - Percent of wing root moment vs. wing station (down bending, D-1)

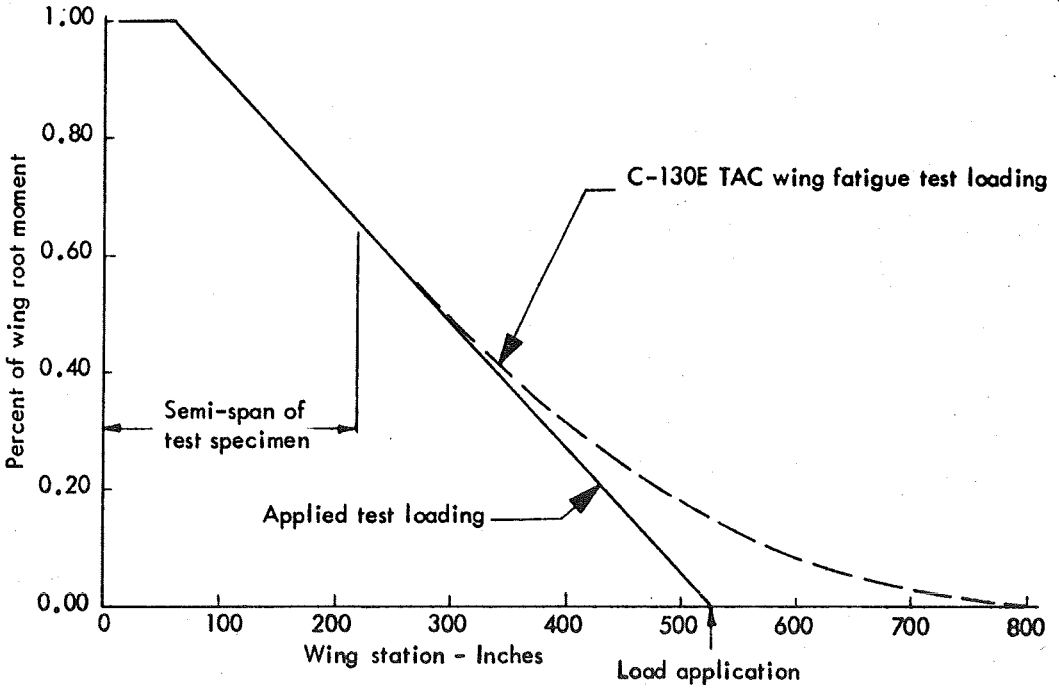
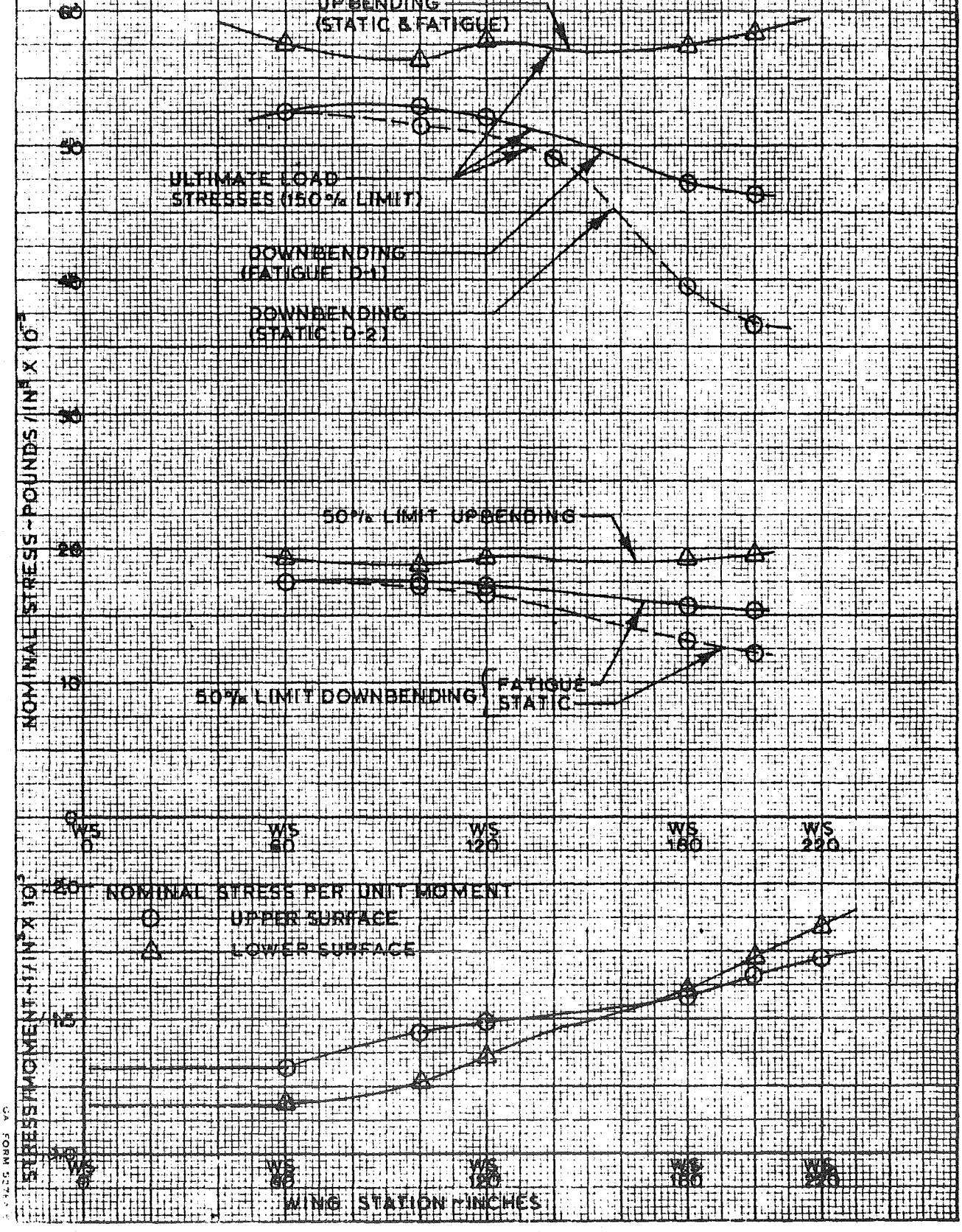
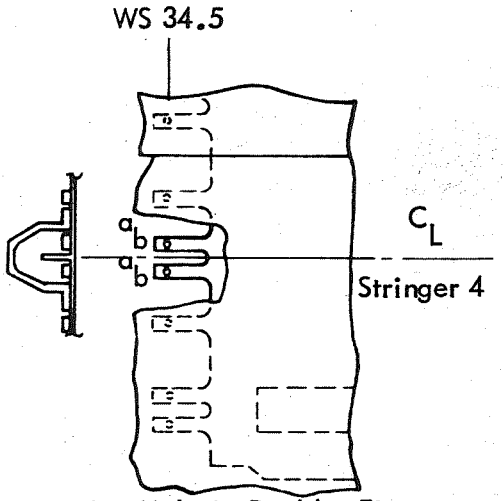


Figure 8. - Percent of wing root moment vs. wing station. up bending

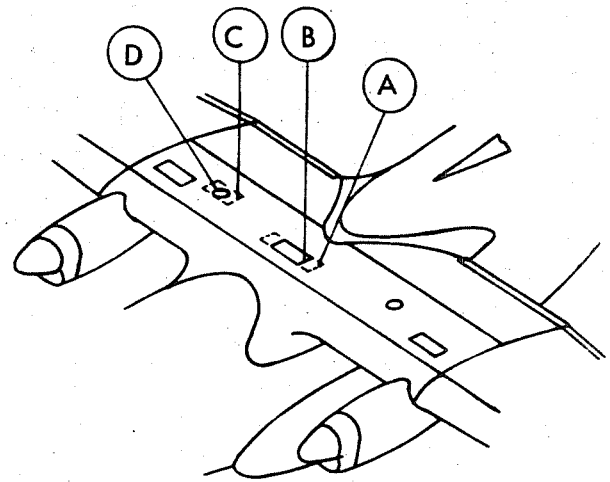
FIGURE 9

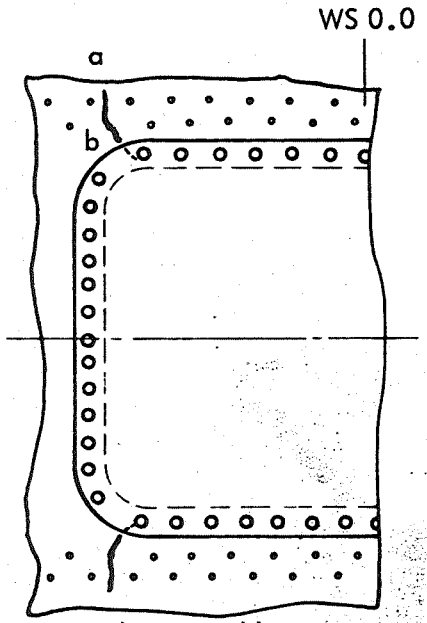
APPROXIMATE WING SKIN TENSION STRESS AND STRESS/MOMENT RATIO VRS. WING STATION - CENTER WING BOX





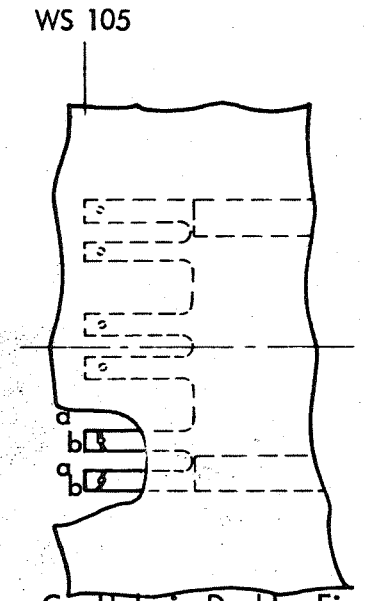
A. Hole in Doubler Finger Under Stringer



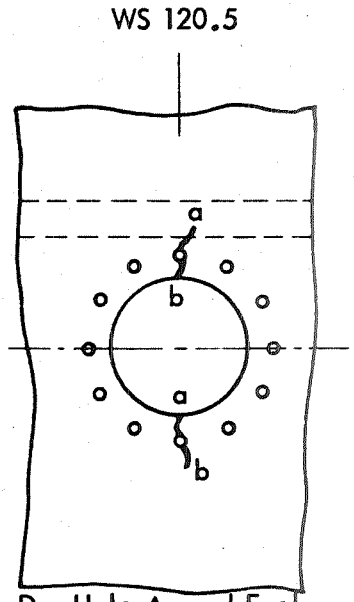


B. Hole in Doubler at Access Door Corner

Fwd



C. Hole in Doubler Finger



D. Hole Around Fuel Filler Cutout

Figure 10 - Typical method of identifying upper surface skin cracks originating in fastener holes.

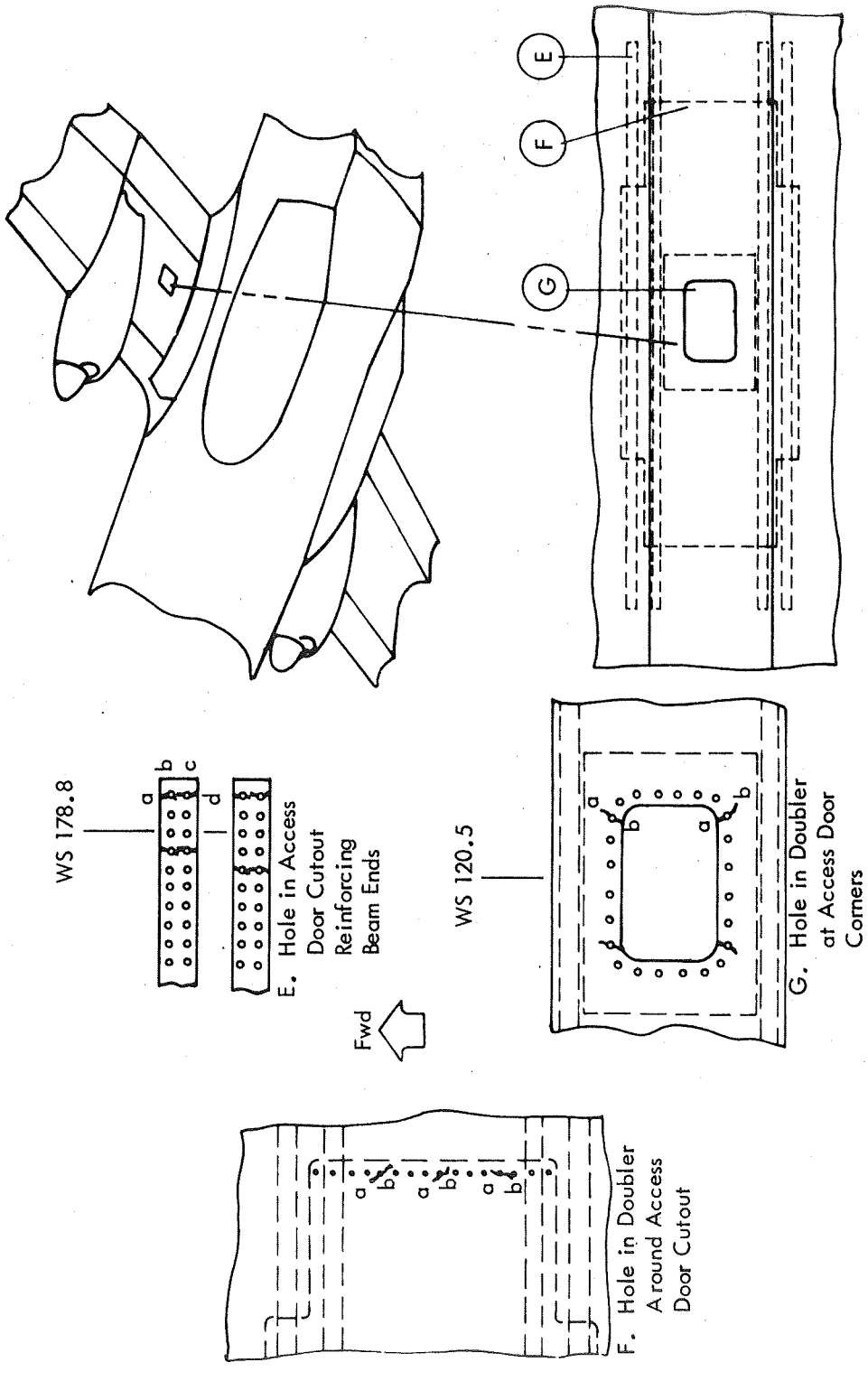


Figure 11 - Typical method of identifying lower surface skin cracks originating in fastener holes.

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TABLE I. TEST EQUIPMENT

Item	Equipment	Manufacturer	Serial No.	Type	Calibration Not Required	*Calibration Void After
1	Digital Strain Indicator	Budd Instruments	SGE 104	30 ch		9-1-71
2	Function Generator		SG-23	Standard	x	
3	Offner Dynograph Recorder	Beckman	REC-85	R		
4	Servac	MTS Research, Inc.	851	Standard	x	
5	Servac	MTS Research, Inc.	693	Standard	x	
6	Servac	MTS Research, Inc.	694	Standard	x	
7	Servac	MTS Research, Inc.	849	Standard	x	
8	Load Cell	BLH	TD 730	100K		9-1-71
9	Load Cell	BLH	TD 728	100K		9-1-71
10	Load Cell	BLH	TD 729	100K		9-1-71
11	Load Cell	BLH	TS 788	100K		9-1-71
12	Oscilloscope	Hewlett Packard	O-46	Model 120B	x	
13	Strain Indicator	BLH	SGE-79	N		3-30-71
14	Balance Box	BLH	SGE-110	225	x	

* Calibrations are directly traceable to the National Bureau of Standards.

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TABLE II

TEST LOADS @ 100% LIMIT LOAD

		WING STATION		
		0	61.6	220
Upbending	Moment	32,437,800 in.lbs.	32,437,800 in.lbs.	21,349,800 in.lbs.
	Shear	0	70,000 lbs.	70,000 lbs
	C_p of Shear	←	37.34% chord F.S. 554.13	→
Downbending (D-1)	Moment	-26,542,494 in.lbs.	-26,542,494 in.lbs.	-17,566,600 in.lbs.
	Shear	0	-56,666 lbs.	-56,666 lbs.
	C_p of Shear	←	40% chord F.S. 559.24	→
Downbending (D-2)	Moment	-27,077,361 in.lbs.	-27,077,361 in.lbs.	-13,033,300 in.lbs.
	Shear	0	-88,662 lbs.	-88,662 lbs.
	C_p of Shear	←	37.49% chord F.S. 554.42	→

TABLE III. NOTES USED THROUGHOUT THE REPORT ON CRACK LENGTH DATA TABLE

(a) All crack length measurements are given to the nearest 0.1 inch.

(b) The crack terminated at the edge of the panel.

(c) The crack terminated in a fastener hole.

(d) See Table B-I for the previous loading history of this specimen.

(e) The crack terminated in a stop-drilled hole.

(f) A W.S.220 Rainbow Fitting node cracked at this time.

(g) Stringers numbered 15, 16, 17 and 18 were also found to be broken at this time.

(h) The crack disappeared under an external repair.

(i) This damage was discovered during an inspection after the residual strength test was completed.

(j) The crack appeared from beneath the opposite side of an external repair.

(k) On Specimen #13, at 19000 cycles, Stringers 17 and 18 were discovered to be partially broken, 17 from its Aft Flange to its centerline and Stringer 18 from its Fwd. Flange to its centerline.

(l) On Specimen 12, broken stringers were found as follows: No. 5 after 2870 cycles, No. 4 after 4034 cycles, Nos. 3, 6, 7, and 8 after 10,500 cycles.

(m) On Specimen #11, during Crack Propagation Test, Phase II, stringers were broken as follows: Stringer 10 @ 13,859 cycles; Stringer 11 @ 14,088 cycles; Stringers 8 & 9 @ 14,197 cycles; and Stringer 7 @ 14,310 cycles.

TABLE IV
SUMMARY OF LOADS

				PROPAGATION TEST						RESIDUAL STRENGTH TEST	
				PHASE 1			PHASE 2				
SPECIMEN NO.	SERIAL NO.	FLIGHT HOURS	MODEL	CYCLES APPLIED	MAGNITUDE, % LIMIT LD.	LOAD CONDITION	CYCLES APPLIED	MAGNITUDE; % LIMIT LD.	LOAD CONDITION	LOAD CONDITION	FAILURE LOAD
1	64-500	5,920	E							Upbending	115.7%
2	63-7838	4,020	↕							↕	114.5%
3	64-514	6,094									115.4%
4	64-539	4,493		7,450	50%-5%	Upbending	3,400	40%-4%	Upbending		64.8%
5	64-534	6,296		8,033	50%-5%	Upbending	11,967	30%-3%	Upbending	Upbending	86.8%
6	62-1828	7,374	E							Downbdg.1	96.3%
7	58-5397	5,617	B							Downbdg.2	97.7%
8	62-1859	7,892	E							↕	116.6%
9	62-1795	11,251	↕	10,000	25%-2.5%	Downbending 1	10,000	22.5%-2.3%	Downbdg.1		81%
10	64-572	6,603		10,000	40%-4%	↕	10,000	35%-3.5%	↕		87%
11	63-9811	7,660		10,000	50%-5%		4,500	40%-4%			57.3%
12	62-1860	5,674		4,034	50%-5%	Downbending 1	6,466	30%-3%	Downbdg.1	Downbdg.2	56%
13	N9268R	13,000	↕	10,000	35%-3.5%	Upbending	10,000	32.5%-3.3%	Upbending	Upbending	62.3%
14	N9259R	13,000	E	6,297	45%-4.5%	Upbending	10,000	22.5%-2.3%	Upbending	Upbending	59%

TABLE V

SUMMARY OF FATIGUE DAMAGE, FAILURE LOADS AND FLIGHT HOURS FOR DOWNBENDING TEST SPECIMENS

Specimen No.	Flight Hours	Description of Area of Crack Origin	Crack No.	Wing Station, In.	Initial Crack Length, In.	Downbending 1, Phase 1 Cyclic Test, R = 0.1			Downbending 1, Phase 2 Cyclic Test, R = 0.1			* Length After Residual Strength Test, In.	% Limit Load at Failure
						Cycles	Max. Load, % Limit Load	Crack Length, In.	Cycles	Max. Load, % Limit Load	Crack Length, In.		
6	7,374	Doubler runout Doubler runout W.S. 220 joint	1 2 9	34.5L 34.5L 220R	4.4 0.3 -							7.0 7.1 Failure line	96.3 ↕ 96.3
7	5,617	Fuel filler cutout Doubler runout Doubler runout Fuel filler cutout	1 3 9 14	120.5L 105.5L 135.3R 120.5R	0.3 2.3 1.4 -							1.6 9.6 Failure line 1.7	97.7 ↕ 97.7
8	7,892	Door cutout corner Door cutout corner W.S. 61 rib cap	1 3 4	199.8L 182.3R 61R	1.2 1.4 -							1.2 1.4 Failure line	116.6 ↕ 116.6
9	11,251	Doubler runout Doubler runout Doubler runout Doubler runout	1 2 3 4	34.5L 34.5L 37.5R 135R	3.6 0.5 0.8 2.8	10,000	25	3.6 0.5 0.8 2.8	10,000	22.5	3.6 0.7 1.3 2.9	Failure line Failure line 19.8 2.9	81 ↕ 81
10	6,603	W.S. 61 rib cap Door cutout corner Door cutout corner Door cutout corner	6 2 1 3	61L 182L 198.5L 199R	- 0.8 0.8 1.5	10,000	40	1.4 1.9 1.1 1.5	10,000	35	1.6 2.5 1.3 1.5	Failure line 3.0 1.4 1.5	87 ↕ 87
11	7,660	W.S. 200 joint Fwd spar cap Door cutout corner	9 3 1	213.5R 178.5R 182.5L	- 0.8 3.0	10,000	50	3.6 1.1 4.0	4,500 4,500	40	17.3 1.7 4.1	Failure line 10.3 4.1	57.3 ↕ 57.3
12	5,674	Doubler runout Fuel filler cutout Fuel filler cutout Doubler runout Fuel filler cutout Fuel filler cutout	3 1 2 8 9 10	105L 120.5L 120.5L 135L 120.5R 120.5R	0.1 0.3 0.3 0.5 0.1 0.3	4,034	50	19.7 1.7 1.6 12.9 1.6 4.2	6,466	30	19.7 1.7 1.6 12.9 1.6 4.2	Failure line 1.7 1.6 12.9 1.6 4.2	56 ↕ 56

* Downbending 2 condition used for all specimens except Number 6.

TABLE VI

SUMMARY OF FATIGUE DAMAGE, FAILURE LOADS AND FLIGHT HOURS FOR UPBENDING TEST SPECIMENS

Specimen No.	Flight Hours	Description of Area of Crack Origin	Crack No.	Wing Station, In.	Initial Crack Length, In.	Upbending, Phase 1 Cyclic Test, R = 0.1			Upbending, Phase 2 Cyclic Test, R = 0.1			* Length After Residual Strength Test, In.	% Limit Load at Failure
						Cycles	Max. Load, % Limit Load	Crack Length, In.	Cycles	Max. Load, % Limit Load	Crack Length, In.		
1	5,920	Beam runout Door cutout corner Door cutout corner	1 2 4	176.5L 113R 113L	3.7 0.3 0.3							Failure line 1.7 1.7	115.7 ↕ 115.7
2	4,020	Beam runout Beam runout Beam runout	1 2 3	176L 176L	3.7 1.0 0.8							Failure line Failure line Failure line	114.5 ↕ 114.5
3	6,094	Beam runout Door cutout corner	2 7	176R 113R	2.0 2.0							Failure line 2.0	115.4 115.4
4	4,493	Beam runout Beam runout Door cutout corner	1 2 5	176L 182L 112.8R	2.6 1.3 0.3	7,450	50	6.1 4.9 3.2	3,400	40	6.8 4.9 3.2	Failure line Failure line 3.2	64.8 ↕ 64.8
5	6,296	Door cutout corner Wing joint bolt node	1 16,17	112.8R 220R	1.1	8,033	50	3.7	11,967	30	4.0	5.5 Failure line	86.8 86.8
13	13,000	Beam runout Drain hole Drainhole Door cutout corner Beam runout Beam runout	3 6 7 13 15 16	181L 179L 173.5L 113L 58L 58L	2.4 0.2 0.6 0.1 3.3 9.4	10,000	35	3.5 0.9 0.8 1.1 3.3 9.4	10,000	32.5	4.9 1.4 1.7 1.6 5.2 16.8	5.2 1.4 1.8 1.6 Failure line Failure line	62.3 ↕ 62.3
14	13,000	Doubler runout W.S. 220 joint Beam runout' Beam runout	15 3 23 27	168L 213.7L 59R 179R	0.1 0.8 0.1 2.5	6,297	45	22.7 1.5 4.4 3.3	10,000	22.5	22.7 2.1 5.0 3.3	23.7 2.1 5.4 Failure line	59 ↕ 59

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TABLE VII. FINAL CUMULATIVE LOWER SURFACE CRACK LENGTHS AT MAJOR WING STATIONS AFTER FINAL FAILURE

SPECIMEN NUMBER	WING STATION, INCHES					CRACK NUMBERS (Reference)	REPAIRS
	58	113	128	168/182	220		
1		1.7R	2.0L	Failure (L)		1,3 2 4,5	None
2		0.3R		Failure (L) 6.8R		1,2,3 6,7,8 5	None
3		2.0R 0.3L	0.3R 1.6L	6.0 Failure (R)		10 1,2,3,4 7 6 8 9	None
4		6.1L 4.6R	1.8R	Failure (L) 18.7R		1,2,4,12 6,9,15,17 7,22 5,11 8	None
5		5.5R	2.5R	34.6L 9.6R	Failure (R)	16,17,18 6,9,12 5 1 4,11	Sling Shot Doublers at W.S. 182 and bushings in access door cutout corners.
13	Failure (L)	3.2R 3.2L	1.6R 3.2L	16.0L 9.3R		15,16 17,18 19 11,12 13,14 3,4,5,6,7,30,40 20,24,39	Sling Shot Doublers at W.S. 182. External Doublers at W.S. 58R. (Repairs removed from W.S. 58L).
14	11.3(L)	0.3L		Failure (R) 38.6 (L)		27,58,60 15,6,5,4,9,10,40,31 23,24 21	Sling Shot Doublers at W.S. 182. External Doublers at W.S.58R . Repairs removed from W.S. 58L.

L = Left
R = Right

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TABLE VIII. FINAL CUMULATIVE UPPER SURFACE CRACK LENGTHS AT MAJOR WING STATION AFTER FINAL FAILURE

SPECIMEN NO.	WING STATION, INCHES									CRACK NUMBER (Reference)	REPAIRS
	34/37	61	105	120.5	135	182	200	213	220		
6	10.7R 14.1L			4.4L 0.6R					Failure (R)	- 8 1,2 5,6 4,7	External doubler around fuel filler cutout.
7			9.6L 2.6L	2.0R 1.6L	Failure (R) 1.5L					7-10 11,12 13,14 1 2,3 4-6	None
8		Failure (R)				0.7L 1.4R	1.2L			4 1 2 3	External doubler repair at doubler runout areas of access door cutout and fuel filler cutouts.
9	Failure (L) 19.8R		1.3R		2.9R 1.1L					1,2 3 5,8,13 4 6,9	None
10		Failure (L) 1.9R				1.7R 3.0L	1.4L 1.5R	0.6L		6,17,19 9,12,21 8 2 1 3 10	Same as Specimen 8.

(CONTINUED)

L = Left
R = Right

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TABLE VIII. FINAL CUMULATIVE UPPER SURFACE CRACK LENGTHS AT MAJOR WING STATION AFTER FINAL FAILURE (CONTINUED)

SPECIMEN NO.	WING STATION, INCHES									CRACK NUMBER (Reference)	REPAIRS
	34/37	61	105	120.5	135	182	200	213	220		
11		2.4L 1.1R				2.7R 5.8L	0.2L 0.3R	Failure (R) 15.8L		9,11 5,34 1,26 22,23,24,62 20,21,59 7,8,31,40,41 6 58	Outboard access door cutout corners (removed prior to test). Center access door doublers and fuel filler doublers.
12	6.1L 7.2R		Failure (L) 11.1R	3.3L 5.8R	12.9L					3,30,32 1,2 8 21,22,27,28,29 11 9,10 16	Doughnut doublers around the fuel filler cutours (removed prior to test).

L = Left
R = Right

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PART 1

Specimen 1

RESIDUAL STRENGTH
UPBENDING TEST RESULTS

TEST RESULTS

The principal initial damage located in the reinforcing beam runout area, was a skin crack (number 1) 3.7 inches long in Panel #2 at W.S. 176.5L, under Stringer 16. This crack extended from the forward edge of the skin panel into a fastener hole common to the aft flange of stringer 16. Crack number 1 originated in the edges of fastener holes common to the reinforcing beam located under stringer 16. Other initial damage consisted of 0.3-inch skin cracks (numbers 2 and 4) in a corner of each access door cutout at W.S. 113.

After applying 87% percent of limit load, skin crack number 3, seven inches long, was found to have originated in the reinforcing beam runout area under Stringer 20 at W.S. 176.5L. No crack extension of crack number 1 under Stringer 16 was noted at that time. At 115.7 percent of limit load, total chordwise failure of the specimen occurred along the reinforcing beam runout cracks located at W.S. 176.5L. Cracks numbers 5 and 6 were found after test completion. Figures 1-1 and 1-2 show the specimen after failure and Table I-1 gives a summary of damage.

TABLE I-1

SUMMARY OF CRACK LENGTHS, SPECIMEN 1

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Upbending Residual Strength Test (d)		
					Initial Crack Length, In.	After Applying 87% Limit Load	After Failure at 115.7% Limit Load
1	The crack originated in the edge of holes common to the W.S. 120 cutout reinforcement beam and the skin	176.5L	0.5" fwd & aft of the stringer 16 ℄	1-1	3.7	3.7	↑
			a	1-2	1.6(b)	1.6	
			d		0.8	0.8	Complete Chordwise Failure
3	Same as crack 1	176.5L	0.5" fwd & aft of the stringer 20 ℄			7.0	↓
			a			2.8	
			d			2.8	
2	The crack originated in the edge of a door attachment fastener hole	113.R	0.8" fwd of the stringer 17 ℄		0.3	0.3	1.7
			a		-		0.9(c)
			b		0.3(b)	0.3	0.3
4	Same as crack 2	113L	0.8" fwd of the stringer 17 ℄		0.3	0.3	1.7
			a		-	-	0.9(c)
			b		0.3(b)	0.3	0.3
5	Same as crack 2	113L	0.8" aft of the stringer 19 ℄				0.3
			a				0.3(b)
6	The crack originated in a drain hole in the skin	174L	2.8" fwd of the stringer 19 ℄				1.4
			a				0.5
			b				0.6(c)

NOTE: Footnotes found in Table III of the General Section, Part G.

PART 2

Specimen 2

RESIDUAL STRENGTH UPBENDING TEST RESULTS

TEST RESULTS

Three initial cracks, numbers 1, 2 and 3, were found at W.S. 176L. Cracks numbers 1 and 3, 3.7 and 0.8 inches long, were located in the reinforcing beam runout areas of Stringers 16 and 20, respectively. Crack number 2, 1.0 inch long, originated in a drain hole located aft of Stringer 15. Final failure occurred along the chordwise row of these cracks.

After applying 100 percent of limit load, crack 3 extended to a total length of 7.5 inches. Subsequently, after applying 110 percent load, crack 3 was found to have extended to a total length of 9.5 inches. Final chordwise failure occurred at 114.5 percent of limit load along the reinforcing beam runout cracks numbered 1, 2 and 3 at W.S. 176.5L. Cracks numbers 4 through 8 were found after test completion. Figures 1 and 2 show the initial and final damage to the specimen in the vicinity of W.S. 176L, and Table I-2 gives a summary of total amount of damage discovered.

TABLE I-2

SUMMARY OF CRACK LENGTHS, SPECIMEN 2

LOWER SURFACE SKIN, (a)

Crack No	Description	Wing Sta. In.	Crack Location	Fig. No.	Initial Crack Length, In.	Upbending Residual Strength Test (d)		
						After Applying 100% Limit Load	After Applying 110% Limit Load	After Failure at 114.5% Limit Load
1	The crack originated in the edges of fasteners common to the W.S. 120 cutout reinforcement beam and the skin	176L	0.5" fwd & aft of the stringer 16 ℄	2-1	3.7	5.0	5.0	
			a		1.6(b)	1.6	1.6	
			b		0.8(c)	2.1	2.1	
2	The crack originated in a drain hole in the skin	176L	2.3" aft of the stringer 15 ℄	2-2	1.0	1.0	1.0	Complete Chordwise Failure
			b		1.0(b)	1.0	1.0	
3	Same as crack 1	176L	0.5" fwd & aft of the stringer 20 ℄		0.8	7.5	9.5	
			a			4.6	6.6	
			b					
			c					
			d			1.5(b)	1.5	
4	Same as crack 2	174L	2.8" fwd of the stringer 15 ℄					1.8
			a					1.0
			b					0.6
5	The crack originated in the edge of a door attachment fastener	113R	0.8" fwd of the stringer 17 ℄					0.3
			b					0.3
6	Same as crack 1	176R	0.5" fwd & aft of the stringer 15 ℄					3.2
			a					1.0(c)
			d					0.9(c)
7	Same as crack 1	176R	0.5" fwd & aft of the stringer 20 ℄					0.4
			d					0.4
8	Same as crack 1	176R	0.5" fwd & aft of the stringer 21 ℄					3.2
			a					1.1(c)
			d					0.9(c)

NOTE: Footnotes found in Table III of the General Section, Part G.

PART 3

Specimen 3

RESIDUAL STRENGTH
UPBENDING TEST RESULTS

TEST RESULTS

Initial damage consisted of six cracks, numbered 1 through 5 and 10, in the reinforcing beam runout area. The longest, crack 2, was two inches in length and was located under stringer 21 at W.S. 177R. Further initial damage consisted of four cracks found in the access door cutout corners, numbered 6 through 9. The longest of these, crack 7, was 2 inches in length, and this crack did not propagate during the subsequent test.

After applying 100 percent of limit load, crack extensions of 2.5 and 1.8 inches of cracks 1 and 2, respectively, were noted in the beam runout area at W.S. 176R. Final chordwise failure occurred along that chordwise station, through cracks 1 through 4, when loaded to 115.4 percent of limit load.

Initial fatigue damage is shown in Figures 3-1 through 3-5. Table I-3 gives a summary of damage.

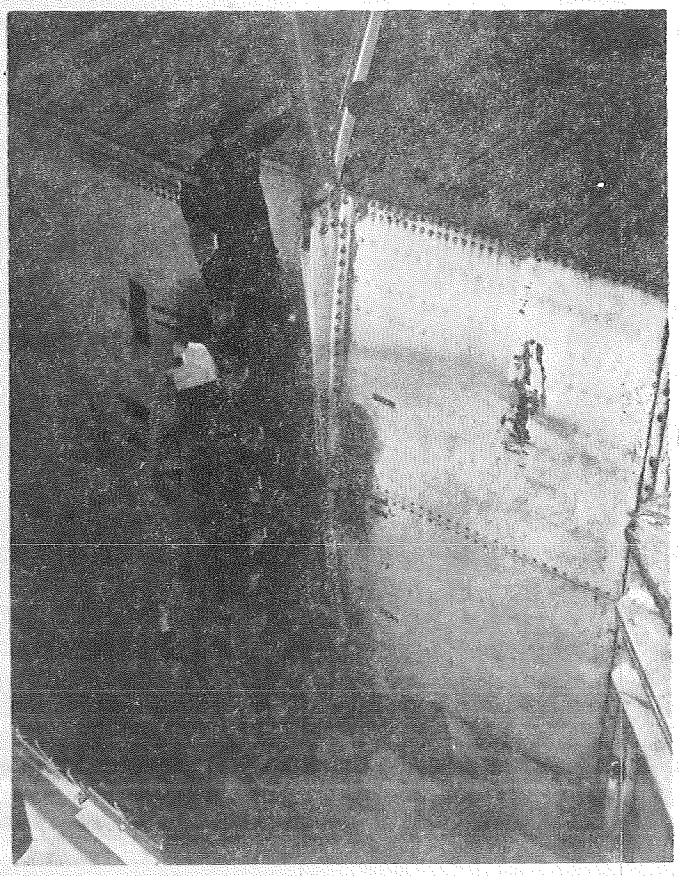
TABLE I-3
SUMMARY OF CRACK LENGTHS, SPECIMEN 3
LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Initial Crack Length, In.	Upbending Residual Strength Test (d)	
						After Applying 100% Limit Load	After Failure at 115.4% Limit Load
1	The crack originated in the edge of fastener holes common to the W.S. 120 cutout reinforcement beam and the skin	176R	0.5" fwd & aft of the stringer 15 ℄	3-1	0.5	3.0	
			a			1.0	
			b				
			c		0.1		
			d		0.1	1.0	
2	Same as crack 1	176R	0.5" fwd & aft of the stringer 16 ℄	3-1	2.0	3.8	Complete Chordwise Failure
			a		0.4	1.4	
			d		0.3	1.4	
3	Same as crack 1	177R	0.5" fwd & aft of the stringer 20 ℄	3-3	0.1	0.1	
			d		0.1	0.1	
4	Same as crack 1	176R	0.5" fwd & aft of the stringer 21 ℄	3-3	0.7	0.7	
			c		0.2	0.2	
			d		0.2	0.2	
5	Same as crack 1	177R	0.5" fwd & aft of the stringer 21 ℄	3-3	0.2	0.2	0.2
			a		0.1	0.1	0.1
			d		0.1	0.1	0.1
6	The crack originated in the edge of an access door attachment fastener hole	128R	0.8" fwd of the stringer 19 ℄	3-4	0.3	0.3	0.3
			a		0.3(b)	0.3	0.3
7	Same as crack 6	113R	0.8" aft of the stringer 17 ℄	3-4	2.0	2.0	2.0
			a		1.2(c)	1.2	1.2
			b		0.3(b)	0.3	0.3
8	Same as crack 6	113L	0.8" aft of the stringer 17 ℄	3-5	0.3	0.3	0.3
			b		0.3(b)	0.3	0.3
9	Same as crack 6	128L	0.8" aft of the stringer 17 ℄	3-5	0.3	0.3	1.6
			a		-	-	0.8
			b		0.3	0.3	0.3
10	Same as crack 1	176L	0.5" fwd & aft of the stringer 16 ℄	3-2	1.5	1.5	6.0
			a		-	-	1.6(b)
			d		0.3	0.3	3.1(c)
11	The crack originated in the edge of a drain hole in the skin	173L	2.8" fwd of the stringer 15 ℄				3.0
			a				0.9(c)
			b				1.9(c)

NOTE: Footnotes found in Table III of the General Section, Part G.

RJ 5161-1

Figure 1-1
Complete chordwise failure at W.S.176.5L
at 115.7% of limit upbending load



RJ 5161-4

Figure 1-2
Complete chordwise failure at W.S.176.5L
at 115.7% of limit upbending load

RJ 5208-3

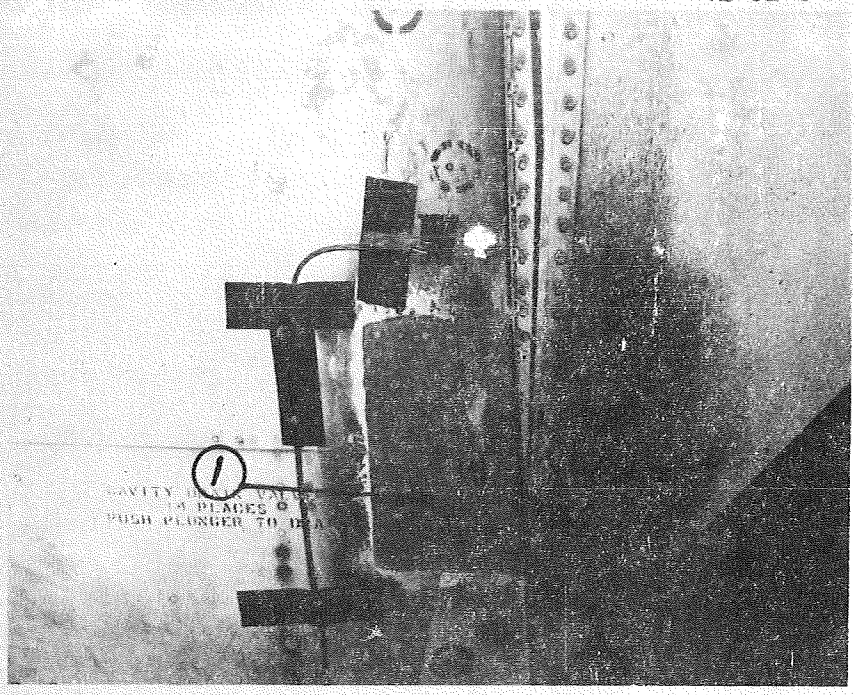
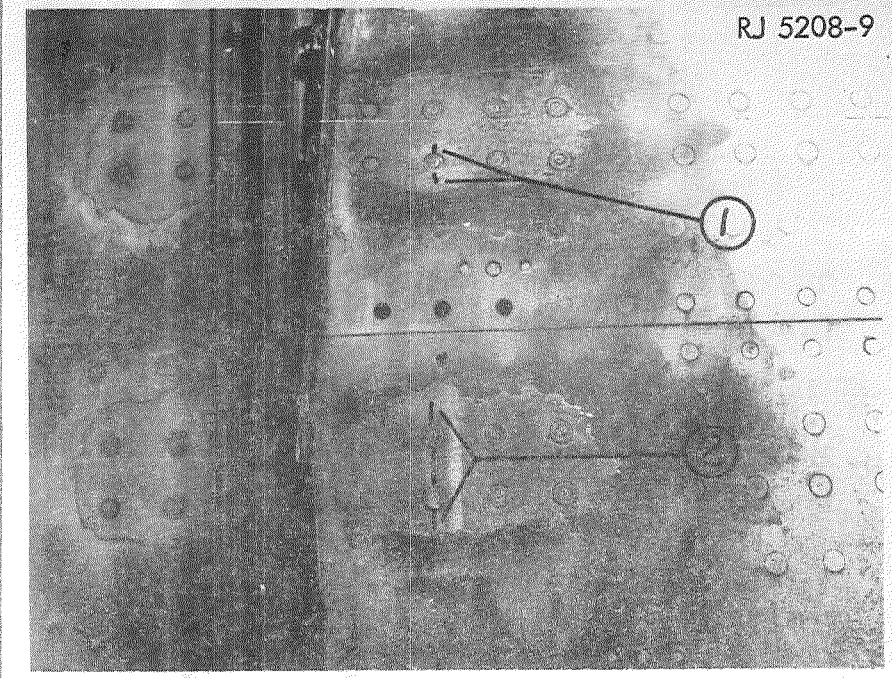
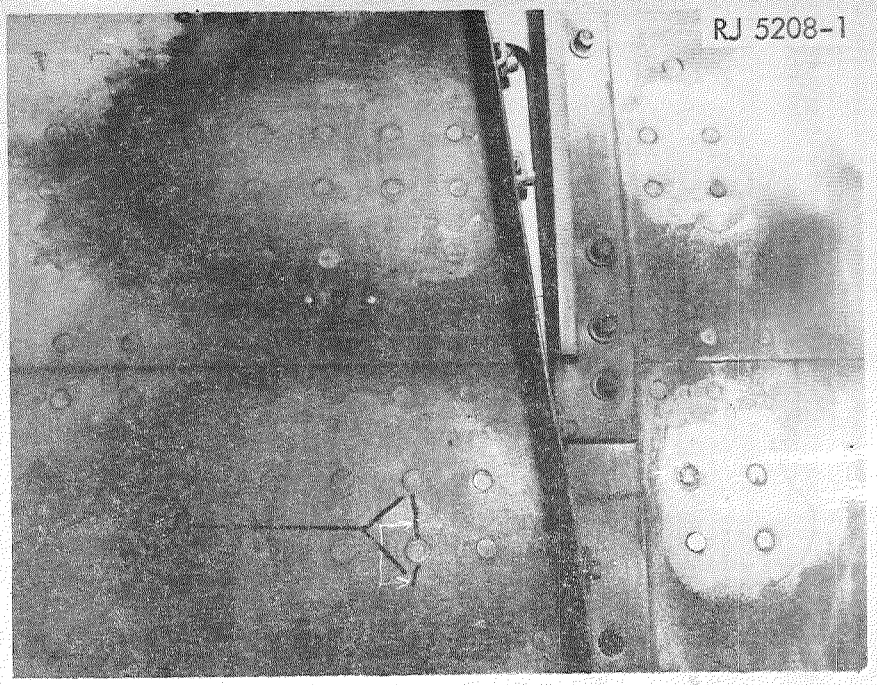


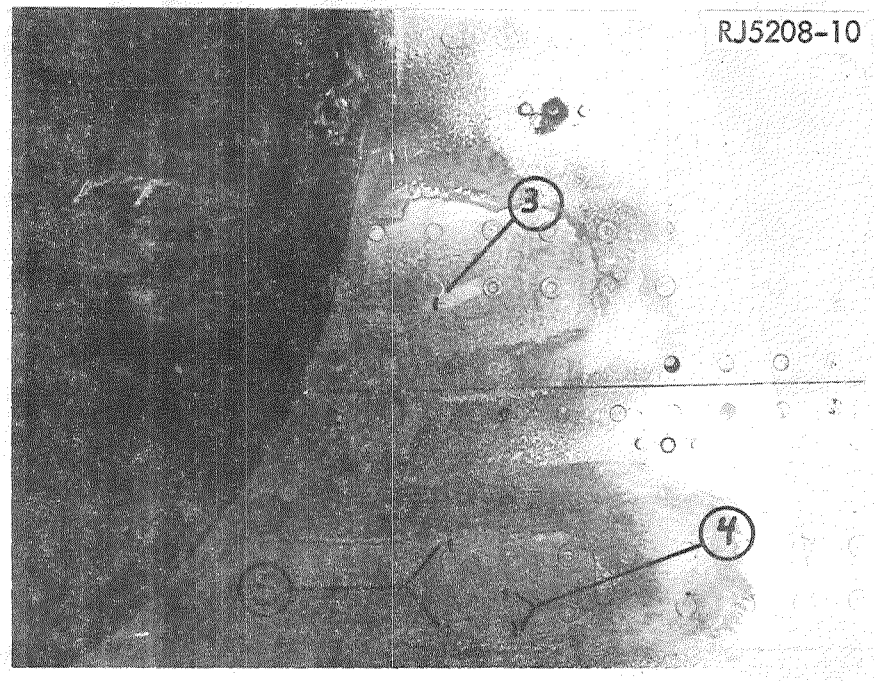
Figure 2-1
Initial Damage
Crack 1 originated in fastener holes common to the skin and reinforcing beam under stringer 16 at WS 176L.

RJ 5208-5

Figure 2-2
Final Failure
The final failure occurred through cracks 1, 2 and 3 along WS 176L at 114.5% of limit load.







Initial skin cracks originating in fastener holes common to the reinforcing beams of the access door cutout. (W.S.176L & R)

Figure 3-1 (Upper Left)
Cracks 1 and 2 under stringers 15 and 16.
(WS 176R)

Figure 3-2 (Upper Right)
Crack 10 under stringer 16 (W.S.176L)

Figure 3-3 (Left)
Cracks 3, 4 & 5 under stringers 20 and 21
(W.S. 176R)

RJ 5208-11

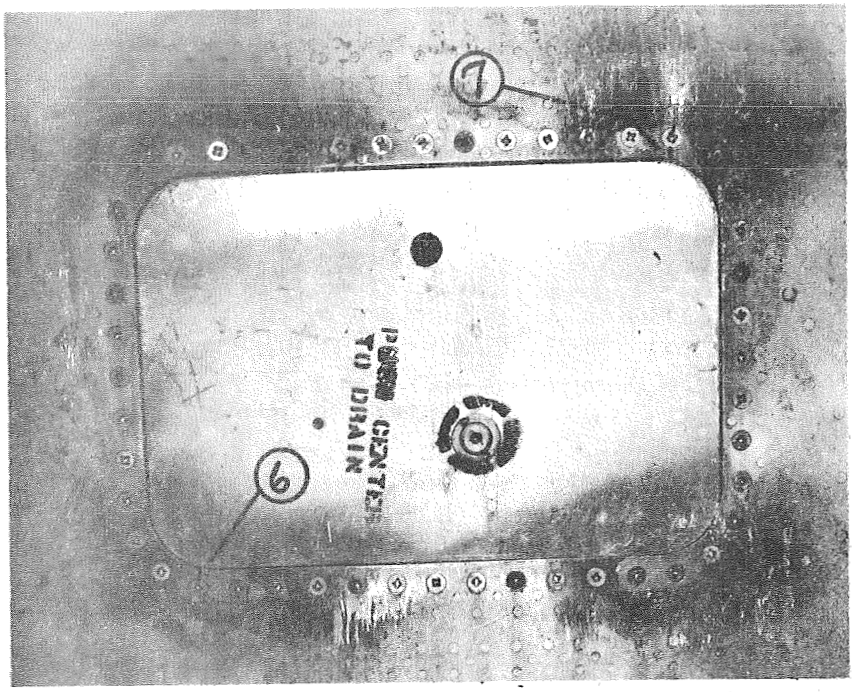


Figure 3-4 Skin cracks 6 and 7 in corners of the right access door cutout.

RJ 5208-12

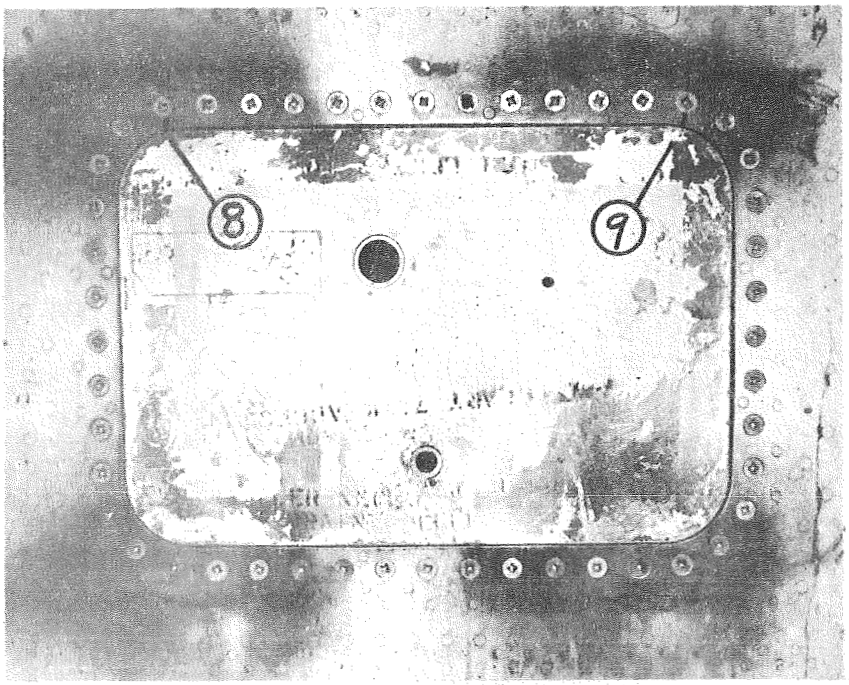


Figure 3-5 Skin cracks 8 and 9 in the corners of the left access door cutout.

PART 4

Specimen 4

CRACK PROPAGATION & RESIDUAL STRENGTH UPBENDING TEST RESULTS

TEST RESULTS

Initial Damage

When received, the specimen had 4,493 flight hours logged and had sustained the following initial fatigue damage:

1. Reinforcing beam runout areas, W.S. 176-179. Five skin cracks, numbers 1, 2, 3, 4 and 6 were found. Crack 1, 2.6 inches long, was the longest single crack. It was located under stringer 16 at W.S. 176L. Total chordwise cracking along cracks 1, 2 and 4 was 4.5 inches and final failure occurred through these cracks. The appearance of cracks 1, 2, 3 and 4 prior to testing is shown in Figures 4-1 and 4-2. Crack 6 is shown in Figure 4-11.

2. Access door cutout corner - W.S. 112.8R. A 0.3-inch-long skin crack, number 5, was detected in the forward inboard corner of the right access door cutout.

Propagation Test

Phase 1

This portion of testing consisted of applying a constant amplitude cyclic load which varied from 50% to 5% of the upbending limit load for 7,450 cycles. Perceptible crack propagation began after a few hundred cycles; most of the initial cracks grew steadily, and several new cracks were initiated. At 7,450 cycles, cracks 1 and 4 in Panels #1 and #2 at W.S. 176L had grown to a total of 10.6 inches; at W.S. 176R, cracks 6, 9, 15 and 17 had grown in Panels #2 and #3 to a total of 12.4 inches.

Phase 2

The cyclic load amplitude was reduced to 40% to 4% of limit load because of the extensive amount of cracking which had occurred. After 3,400 additional cycles, cumulative crack lengths of 19 inches at W.S. 176L and 14.5 inches at W.S. 168L had been developed and cyclic testing was terminated. Damage after crack propagation testing is shown in Figures 4-3 and 4-4 for W.S. 176L, and in 4-5 for W.S. 168L. There was also a 17.6-inch crack accumulation at W.S. 176R. Damage at W.S. 176R after crack propagation testing is shown in Figures 4-12 and 4-13. Cyclic testing was discontinued because damage at this point was considered sufficiently extensive to risk an uncontrolled total failure of the specimen at any time.

TEST RESULTS (Continued)

Static Residual Strength Test

Static residual strength testing was then conducted under the customary control procedures, and progressed uneventfully until a small sonic report was heard at 50% limit load. Failure occurred at 64.8% limit load along an irregular line through W.S. 176L and W.S. 168L. External views of the failure area appear in Figures 4-6 through 4-9. An interior view of the failed panels and stringers appears in Figure 4-10. Figures 4-12 and 4-13 show final cracking at W.S. 176R. Door damage at W.S. 120L is shown in Figures 4-14 through 4-17. At the W.S. 120R door, damage is shown in Figures 4-18 and 4-19 (before testing) and Figures 4-20, 4-21 and 4-22 (after testing).

Table I-4 gives a summary of major fatigue damage accumulated by the specimen and Table II-4 gives a summary of minor fatigue damage.

TABLE I-4

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 4

LOWER SURFACE SKIN

Crack No.	Description	Wing Sta. In.	Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d) 0	500	1000	1500	2000	2500	3000	3500	4000
19	The crack originated in the edge of fastener holes common to the skin and the W.S. 120L door cutout reinforcement beam	181L	0.5" fwd & aft of the stringer 15 ℄	4-3									
			a	4-6									
			d	4-7									
				4-8									
				4-9									
13	Same as crack 19	180L	0.5" fwd & aft of the stringer 16 ℄	4-3									
			a	4-6									
			b	4-7									
			c	4-9									
1	Same as crack 19	176L	0.5" fwd & aft of the stringer 16 ℄	4-7,4-1	2.6	2.9	2.9	2.9	3.7	3.7	3.7	3.7	4.5
			a	4-8,4-3	0.8	1.0(c)	1.0	1.0	1.6(b)	1.6	1.6	1.6	1.6
			d	4-9,4-6	0.5	0.6	0.6	0.6	0.8(c)	0.8	0.8	0.8	1.6
12	The crack originated in the edge of the last fastener hole common to the access door cutout doubler and the skin	168L	At the stringer 17 ℄	4-5									
			a	4-6									
			b	4-7									
				4-8									
				4-9									
29	Same as crack 12	168L	At the stringer 18 ℄	Same as 12									
			a										
			b										
27	Same as crack 12	168L	2.8" aft of the stringer 18 ℄	Same as 12									
			a										
			b										
28	Same as crack 12	168L	At the stringer 19 ℄	Same as 12									
			a										
			b										
14	The crack originated in the edge of a drain hole in the skin	175L	2.7" aft of the stringer 19 ℄	4-4,4-6									
			a	4-7,4-8									
			b										
2	Same as crack 19	182L	0.5" fwd & aft of the stringer 20 ℄	4-2	1.3	1.4	2.2 →	2.7	3.2	3.8			3.8
			a	4-4	0.1	0.1	0.5	0.7	0.9(c)	0.9			0.9
			b	4-6	0.2	0.2	0.3●						
			c	4-7	0.2	0.2	0.3●						
			d	4-8	0.2	0.2	0.5	0.8	1.0	1.5(b)			1.5
4	Same as crack 19	176L	0.5" fwd & aft of the stringer 21 ℄	4-2	0.6	0.9	1.6	2.1 →	2.5	3.0			3.0
			a	4-4	0.1	0.1	0.2	0.4	0.5	0.9(c)			0.9
			b	4-6	0.2	0.2	0.4	0.4●					
			c	4-7				0.4●					
			d	4-8				0.4	0.7	0.9			0.9

NOTE: Footnotes found in Table III of the General Section, Part G.

GA FORM 425A-1

TABLE I-4 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 4

LOWER SURFACE SKIN

Crack No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d)					Upbending, Phase 2 Cyclic Test, R = 0.1 Max. Load = 40% Limit Load — Cumulative Number of Cycles (d)								Length After Failure at 64.8% Limit Load
	4500	5000	5500	6000	6500	7000	7450	8000	8500	9000	9500	10,000	10,500	10,850	
19						2.3	3.2	3.2	3.5	4.0	4.3	4.9	5.8	6.9	
a						0.6	1.1	1.1	1.3	1.5	1.6	1.9	2.5	2.9	
d						0.4	0.8	0.8	0.9	1.2	1.3	1.7	2.1	2.7	
13		0.8	0.8	1.1	1.9 →	2.3	2.3	2.3	2.4	2.6	2.6	2.8	3.0	3.2	
a					0.2	0.4	0.4	0.4	0.4	0.6	0.6	0.7	0.9	1.0	
b				0.1	0.2										
c		0.3	0.3	0.4	0.6										
d		0.2	0.2	0.3	0.7	0.7	0.7	0.7	0.7	0.7	0.7	0.9	0.9	1.0	
1	4.8	5.5	6.1 —								6.1	6.3	6.7	6.8	
a	1.6	1.6	1.6 —											1.6	
d	2.0	2.7	3.3(c) —								3.3	3.5	3.9	4.0	
12		0.8	1.0	1.0	1.3	1.8	2.7	3.0	3.3	3.9	4.2	4.9	9.5	14.5	
a		0.4	0.4	0.4	0.6	0.8	1.3	1.4	1.5(c)	1.5	1.6	1.6	1.6	3.3	
b		0.2	0.3	0.3	0.5	0.7	1.1	1.3	1.5(c)	2.1	2.4	3.1	7.7 →	11.0	Complete Chordwise Failure
29												0.2	0.2		
a															
b												0.2	0.2		
27												2.7	3.2		
a												0.2	0.5		
b												0.1	0.3		
28												0.3	0.3	1.2	
a														0.4	
b												0.3	0.3	0.4	
14		1.2	1.7	1.9	2.2 —									2.2	
a		0.6	0.8	1.0(c)	1.0 —									1.0	
b		0.4	0.6	0.7	1.0(c) —									1.0	
2	3.8	4.2	4.6	4.7	4.8	4.9 —								4.9	
a	0.9	1.3	1.7	1.8	1.9	2.0(c) —								2.0	
b															
c															
d	1.5	1.5	1.5	1.5	1.5	1.5 —								1.5	
4	3.0			3.0	3.6	3.7	4.5	4.7	5.2	5.3	5.5	5.7	5.9	7.3	
a	0.9			0.9	0.9	0.9	1.4	1.4	1.6	1.6	1.8	2.0	2.1	2.3	
b															
c															
d	0.9			0.9	1.5	1.7	2.0	2.3	2.3	2.4	2.5	2.5	2.6	3.8	

NOTE: Footnotes found in Table III of the General Section, Part G.

GA FORM 425A-1

TABLE I-4 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 4

LOWER SURFACE SKIN

Crack No.	Description	Wing Sta. In.	Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d) 0	500	1000	1500	2000	2500	3000	3500	4000
9	Same as crack 19	176R	0.5" fwd & aft of the stringer 15 ℄	4-12				0.2	0.8	1.0	1.8	1.9	2.7
			a						0.2	0.4	0.4	0.4	0.8
			b					0.2	0.3	0.4	0.6		
			c								0.1		
			d								0.1	0.2	0.6
6	Same as crack 19	176R	0.5" fwd & aft of the stringer 16 ℄	4-11	0.7	1.1	2.0	2.3	2.7	3.1	3.1	3.1	4.1
			a	4-12			0.1	0.2	0.6	1.0(c)	1.0	1.0	1.6(b)
			b		0.2	0.4							
			c		0.1	0.2							
			d		0.1	0.1	0.5	0.7	0.8(c)	0.8	0.8	0.8	1.2
15	Same as crack 19	176R	0.5" fwd & aft of the stringer 20 ℄	4-13									
			a										
			b										
			c										
			d										
17	Same as crack 19	176R	0.5" fwd & aft of the stringer 21 ℄	4-13									
			a										
			b										
			c										
			d										
16	Same as crack 14	179R	2.7" aft of the stringer 19 ℄	4-13									
21	Same as crack 19	181R	0.5" fwd & aft of the stringer 15 ℄	4-12									
			a										
			b										
			c										
			d										
20	Same as crack 19	181R	0.5" fwd & aft of the stringer 16 ℄	4-12									
			a										
			b										
			c										
			d										
7	The crack originated in the edge of a door attachment fastener hole	112.8L	0.8" fwd of the stringer 17 ℄	4-14		0.3	0.3	1.0	1.3	1.5	1.6	1.6	1.6
			a	4-16				0.2	0.5	0.7	0.8(c)	0.8	0.8
			b			0.3(b)					0.3	0.3	0.3

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-4 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 4

LOWER SURFACE SKIN

Crack No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d)							Upbending, Phase 2 Cyclic Test, R = 0.1 Max. Load = 40% Limit Load — Cumulative Number of Cycles (d)							Length After Failure at 64.8% Limit Load
	4500	5000	5500	6000	6500	7000	7450	8000	8500	9000	9500	10,000	10,500	10,850	
9	3.1	3.3	3.5	3.7	3.8	4.2	4.2	4.9	5.7	6.0	6.2	6.3	6.4	6.6	7.0
a	1.0	1.3	1.5	1.6	1.7	2.2	2.2	2.9	3.2	3.4	3.6	3.6	3.6	3.6	3.8(c)
b															
c															
d	0.7	0.7(c)	0.7	0.7	0.7	0.7	0.7	0.7	1.2	1.3	1.3	1.4	1.5	1.7	1.8
6	4.1	4.5	4.7	5.2	5.7	5.9	5.9	5.9	5.9	6.0	6.0	6.0	6.0	6.0	6.5
a	1.6	—	—	—	—	—	—	—	—	—	—	—	—	—	1.6
b															
c															
d	1.2	1.6	1.8	2.2	2.7	2.9(c)	2.9	2.9	2.9	3.0	3.0	3.0	3.0	3.0	3.5
15			0.1	0.2	0.7	0.7	0.8	0.9	0.9	1.5	2.0	2.1	2.3	2.5	2.6
a			0.1	0.2	0.2	0.2	0.3	0.3	0.3	0.3	0.4	0.5	0.7	0.7	0.7
b					0.1	0.1	0.2	0.2	0.3	0.3	0.4				
c										0.1	0.4				
d										0.1	0.2	0.3	0.4	0.5	0.6
17						0.2	1.8	1.8	2.0	2.0	2.2	2.3	2.5	2.5	2.6
a							0.2	0.2	0.4	0.4	0.4	0.5	0.6	0.6	0.6
b						0.2	0.3	0.3	0.3						
c							0.4	0.4	0.4						
d							0.3	0.3	0.3	0.3	0.4	0.5	0.6	0.6	0.7
16		0.1	0.4	0.9	0.9	1.3	1.6	1.7	1.8	—	—	—	—	—	1.8
a			0.1	0.5	0.5	0.5	0.7	0.8	0.9(c)	—	—	—	—	—	0.9
b		0.1	0.1	0.1	0.1	0.5	0.6(c)	—	—	—	—	—	—	—	0.6
21						0.9	2.0	2.4	2.6	2.8	3.0	3.2	3.4	3.4	3.4
a							0.4	0.5	0.6	0.7	0.7	1.0	1.2	1.2	1.2
b						0.1	0.2	0.4							
c						0.4	0.5	0.5							
d						0.2	0.2	0.5	0.7	0.8	1.0(c)	1.0	1.0	1.0	1.0
20						0.6	0.9	0.9	1.3	1.3	1.3	1.3	1.4	1.4	1.4
a						0.2	0.2	0.2	0.3	0.3	0.3	0.3	0.3	0.3	0.3
b						0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.3	0.3	0.3
c							0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2
d									0.1	0.1	0.1	0.1	0.1	0.1	0.1
7	1.6		1.6	2.5	2.9	3.2	—	—	—	—	3.2	3.7	4.3	4.9	4.9
a	0.8		0.8	1.7	2.1	2.4(c)	—	—	—	—	2.4	2.9	3.5	4.1	4.1
b	0.3		0.3	0.3	0.3	0.3	—	—	—	—	0.3	0.3	0.3	0.3	0.3

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-4 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 4

LOWER SURFACE SKIN

Crack No.	Description	In.	Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d) 0	500	1000	1500	2000	2500	3000	3500	4000
22	Same as crack 7	112.8L	0.8" aft of the stringer 19 ℄	4-14									
			a	4-15									
			b										
5	Same as crack 7	112.8R	0.8" fwd of the stringer 17 ℄	4-18	0.3	1.1	1.5	———	———	———	———	———	1.5
			a	4-19		0.3	0.7(c)	———	———	———	———	———	0.7
			b	4-20	0.3(b)	0.3	0.3	———	———	———	———	———	0.3
11	Same as crack 7	112.8R	0.8" aft of the stringer 19 ℄	4-19								0.3	1.0
			a	4-21								0.3(b)	0.3
			b										0.2
8	Same as crack 7	128.2R	0.8" aft of the stringer 19 ℄	4-19		0.2	0.2	0.2	0.3	1.1	1.1	1.4	1.7
			a	4-22		0.2	0.2	0.2	0.3(b)	0.3	0.3	0.3	0.3
			b							0.3	0.3	0.6	0.9

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-4 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 4

LOWER SURFACE SKIN

Crack No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d) 4500	5000	5500	6000	6500	7000	7450	Upbending, Phase Cyclic Test, R = 0.1 Max. Load = 40% Limit Load — Cumulative Number of Cycles (d) 8000	8500	9000	9500	10,000	10,500	10,850	Length After Failure at 64.8% Limit Load
22						0.2	0.3					0.3	1.1	1.2	1.2
a						0.2	0.3(b)					0.3	0.3	0.3	0.3
b													0.3	0.4	0.4
5	1.5	1.5	2.1	2.4	2.7	2.8	3.2								3.2
a	0.7	0.7	1.3	1.6	1.9	2.0	2.4(c)								2.4
b	0.3	0.3	0.3	0.3	0.3	0.3	0.3								0.3
11	1.1	1.1	1.3	1.4											1.4
a	0.3	0.3	0.3	0.3											0.3
b	0.3	0.3	0.5	0.6(c)											0.6
8	1.8														1.8
a	0.3														0.3
b	1.0(c)														1.0

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-4

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 4

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered (Inches)	(d) Length After 10,850 Cycles	Length After Failure at 64.8% Limit Load
3	The crack originated in the edge of a fastener common to the skin and the W.S. 120 cutout reinforcement beam and ends in a blind Taper-lok	179L	0.5" fwd of the stringer 20 ℄	4-2	0	0.2	0.2	0.2
			a	4-4		0.2(c)	0.2	0.2
				4-6				
				4-7				
				4-8				
10	The crack originated in the edge of fasteners common to the W.S. 120 cutout reinforcement beams and the skin	180L	0.5" fwd & aft of the stringer 20 ℄	4-4	2350	0.1	1.0	1.0
			a	4-6				
			b	4-7				
			c	4-8		0.1	0.4	0.4
			d				0.3	0.4
23	The crack originated in a door attachment fastener	128L	0.8" aft of the stringer 19 ℄	4-14	7200	0.1	0.1	0.2
			a	4-17		0.1	0.1	0.2
			b					
24	Same as crack 10	181R	0.5" fwd & aft of the stringer 21 ℄	4-13	8550	0.1	0.1	0.1
			a					
			b					
			c			0.1	0.1	0.1
			d					
25	The crack originated in a drain hole in the skin	175R	2.8" aft of the stringer 15 ℄	4-12	8800	0.6	1.1	1.2
			a			0.2	0.4	0.5
			b			0.2	0.5	0.5
26	The crack originated in Panel No. 1 in the edge of a fastener common to the skin and the panel splice (in the flange of stringer 16)	175R	3.4" aft of the stringer 15 ℄	4-12	9850	0.1	0.1	0.3
			a					
			b			0.1	0.1	0.3(b)
30	The crack originated in the edge of a fastener common to the skin and the W.S. 120 cutout reinforcement beam	177R	0.5" fwd of the stringer 15 ℄	4-12	After Failure	0.1		0.1
			a					
			b			0.1		0.1
18	Same as crack 10	181L	0.5" fwd of the stringer 20 ℄		6950	0.6	0.9	0.9
			a			0.2	0.3	0.3
			b			0.1	0.2	0.2
			c			-	0.1	0.1

NOTE: Footnotes found in Table III of the General Section, Part G.

RJ 5281-2

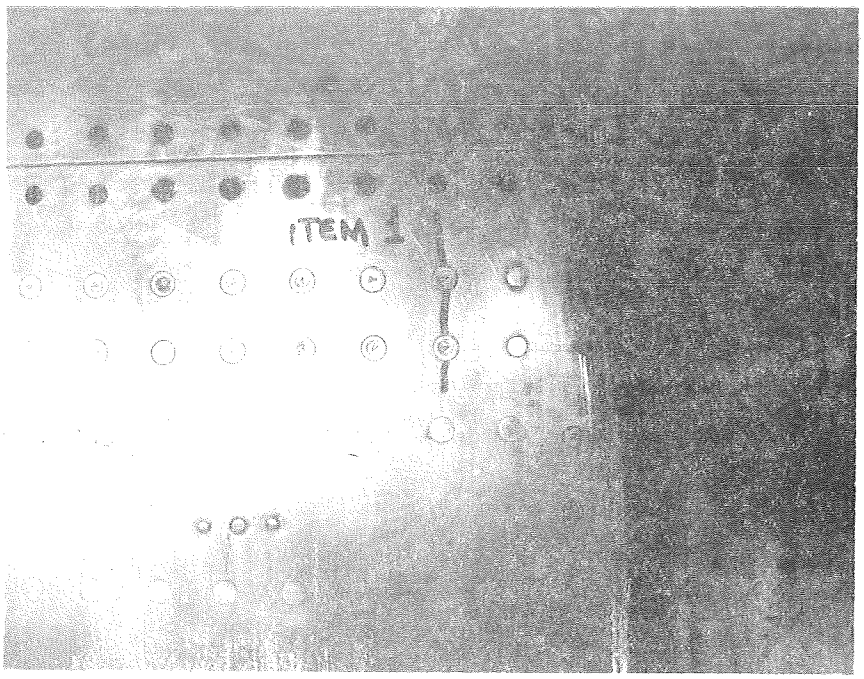


Figure 4-1 WS 176L, stringer 16 area, prior to testing, showing item 1.

RJ 5281-1

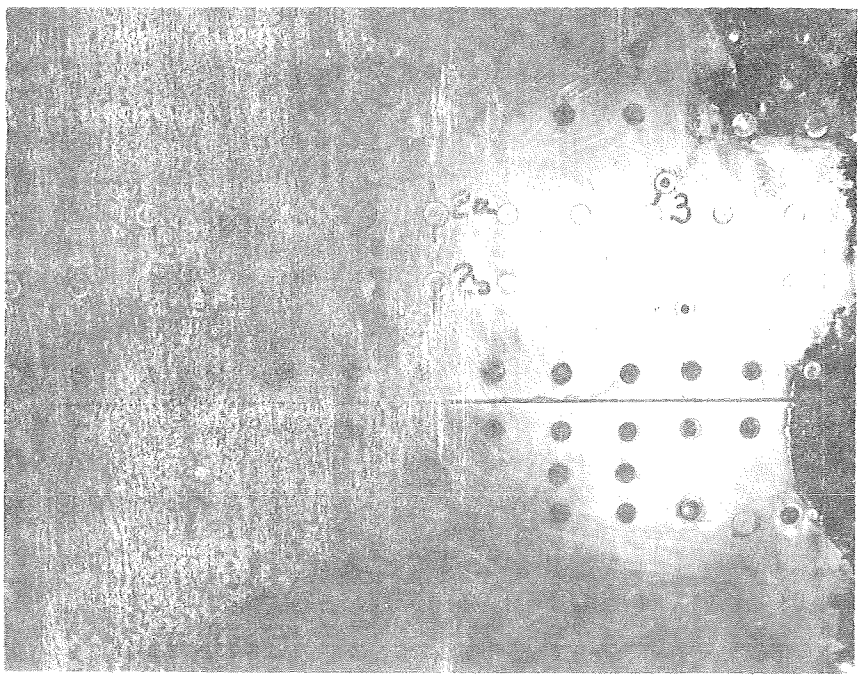


Figure 4-2 WS 176L stringers 20, 21 area, prior to testing, showing items 2 & 4.

RJ 5358-2

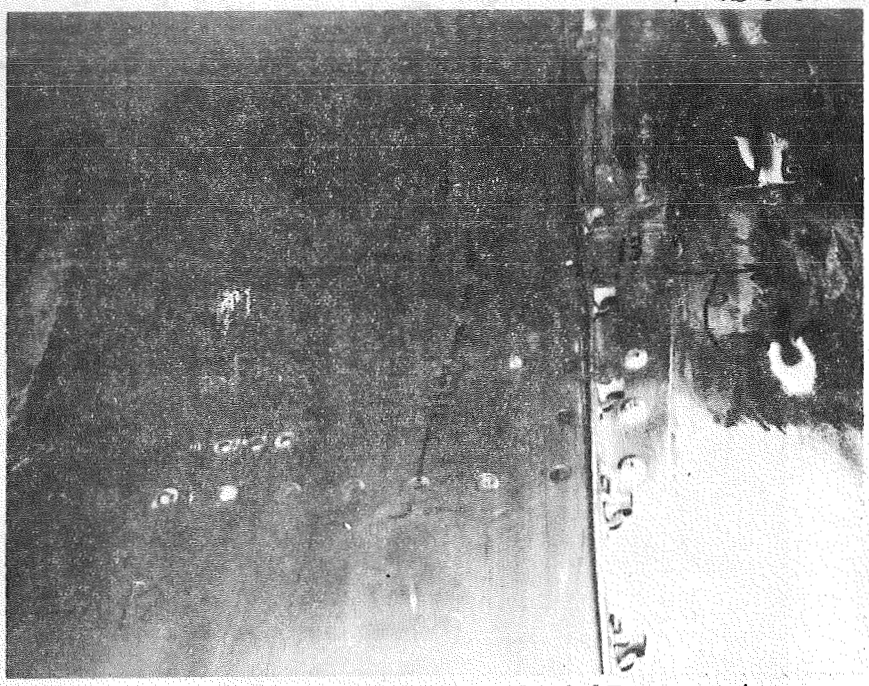


Figure 4-3 WS 176L, stringer 15, 16 & 17 area, subsequent to crack propagation testing, showing items 1, 19 & 13.

RJ 5358-1

Figure 4-4 WS 176L, stringer 19, 20, & 21 area subsequent to crack propagation testing, showing items 28, 14, 3, 18, 10 and 4.

RJ 5358-3

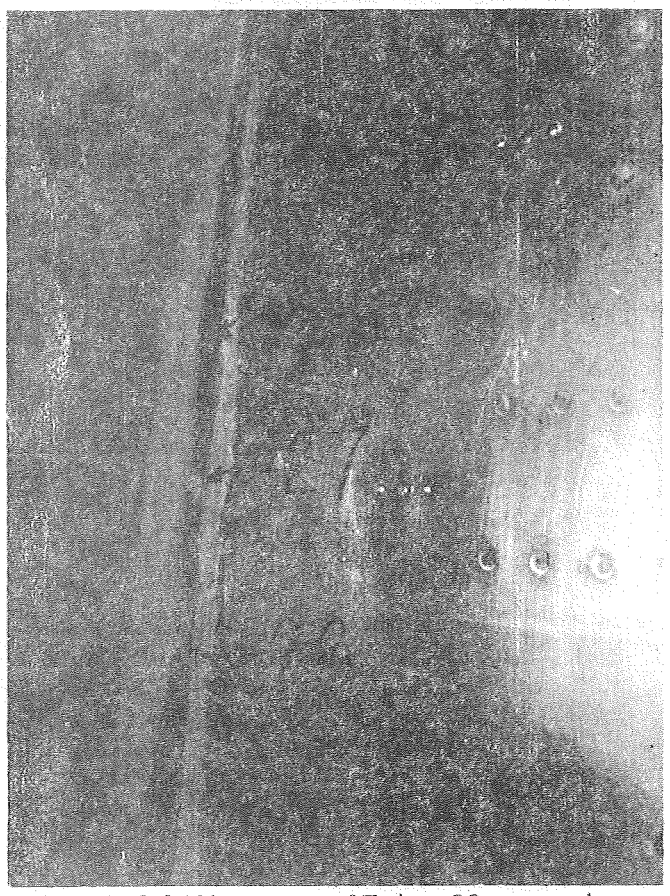
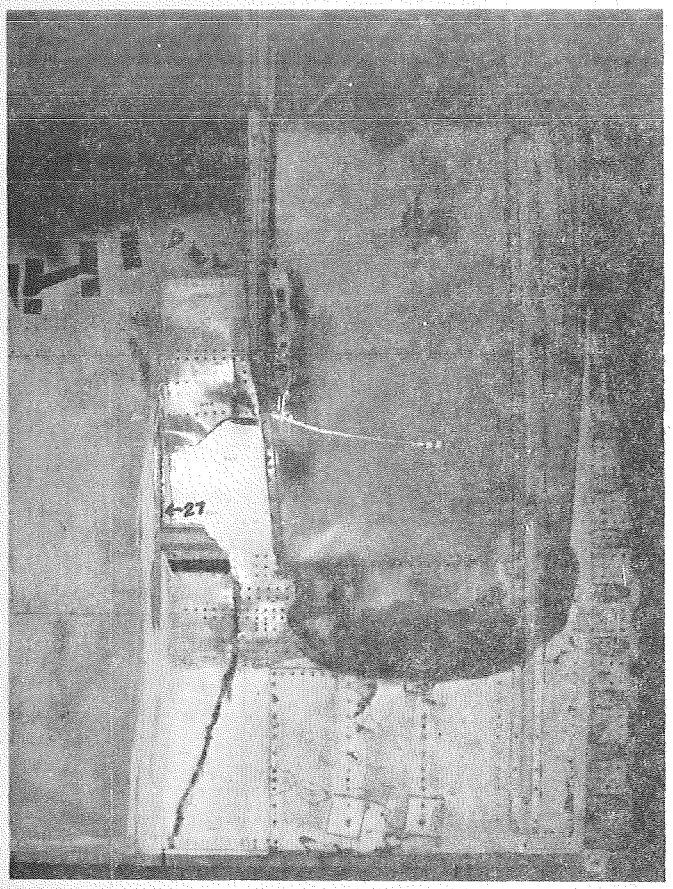


Figure 4-5 WS 168L stringer 17 thru 20 area subsequent to crack propagation testing showing items 12, 29, 27, & 28.



RJ 5366-3

Figure 4-6 (Left)

WS 168L, 176L area subsequent to failure showing items 19, 13, 1, 12, 29, 27, 14, 2, 3, 18 & 4.

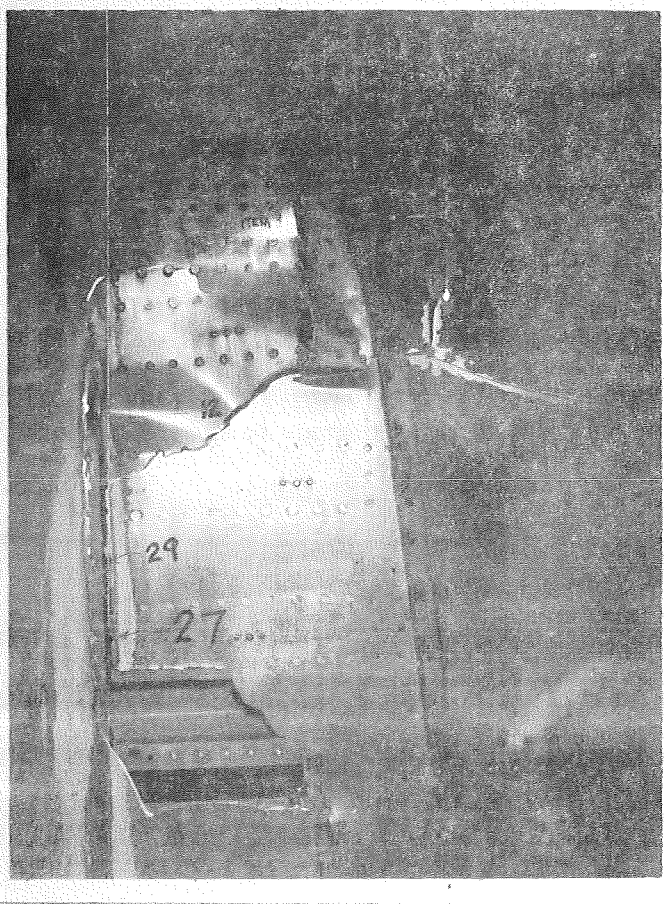
Figure 4-7 (Lower Left)

WS 168, 176L, stringers 15 thru 20 area subsequent to failure showing items 19, 13, 1, 12, 29, 27, 14, 2, 3 & 18.

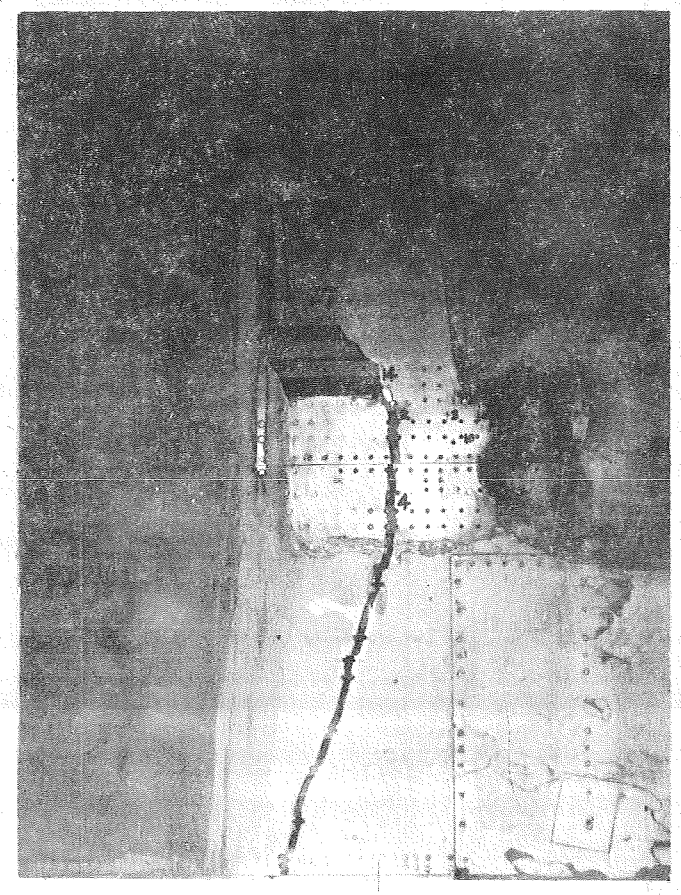
Figure 4-8 (Lower Right)

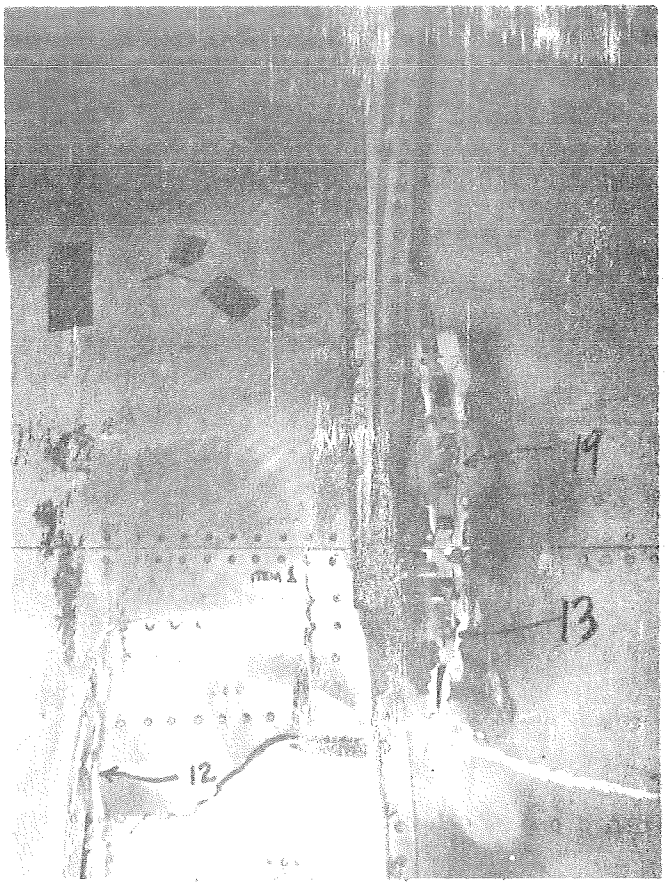
WS 168, 176L, stringers 15 thru 24 area, subsequent to failure, showing items 19, 13, 1, 12, 29, 27, 14, 2, 3, 18, & 4.

RJ 5366-9



RJ 5366-2

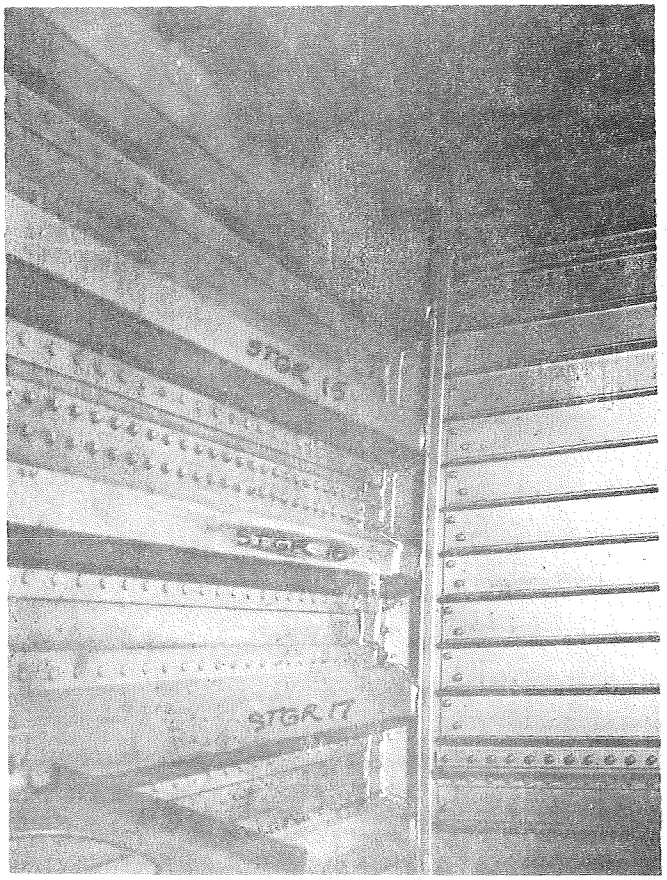




RJ 5366-

Figure 4-9

WS 168-176L, stringers 12 thru 17 area, subsequent to failure, showing items 19, 13, 1 & 72.



RJ 5366-8

Figure 4-10

Internal view WS 176-182L, stringers 12 thru 18 area, showing damage to stringers.

RJ 5281-6

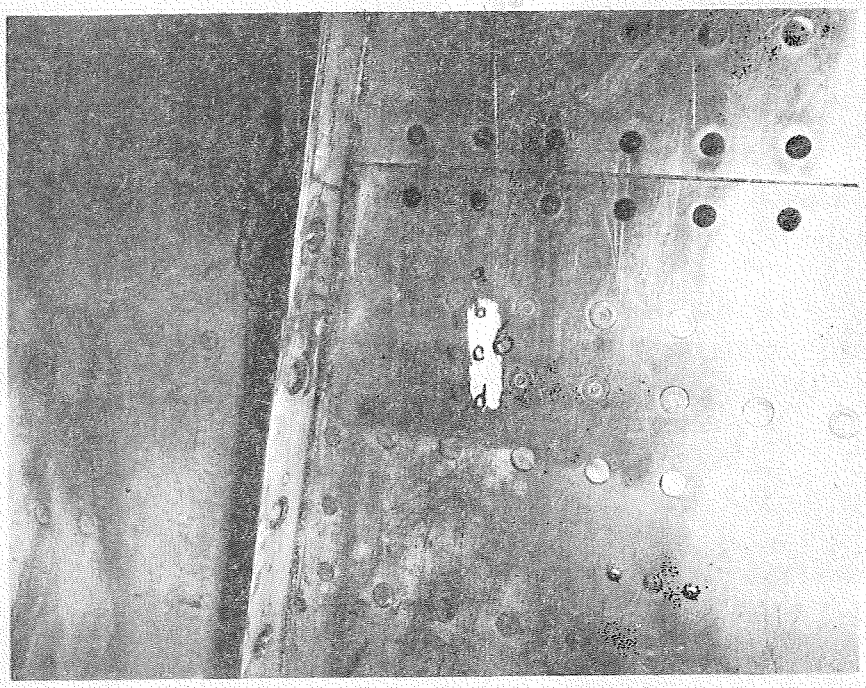
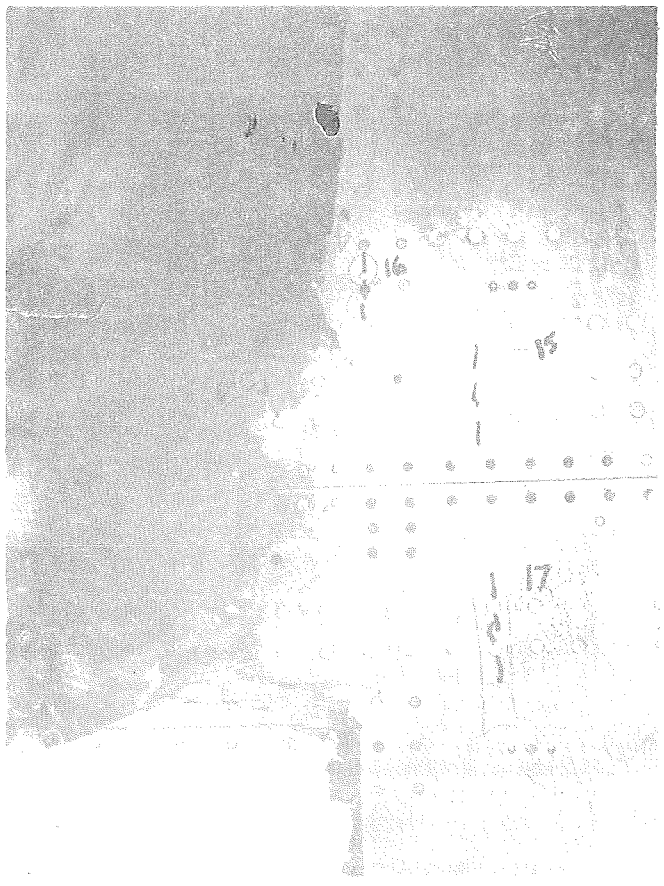


Figure 4-11 WS 176-182R, stringers 14 & 15 area. Prior to testing, showing item 6.

RJ 5366-6

Figure 4-12

WS 168-182R, stringer 14 thru 17, subsequent to failure, showing items 21, 20, 30, 9, 25, 26 & 6.



RJ 5366-7

Figure 4-13

WS 168-182R, stringers 19 thru 22, subsequent to failure, showing items 16, 15, 24 & 17.

RJ 5366-4

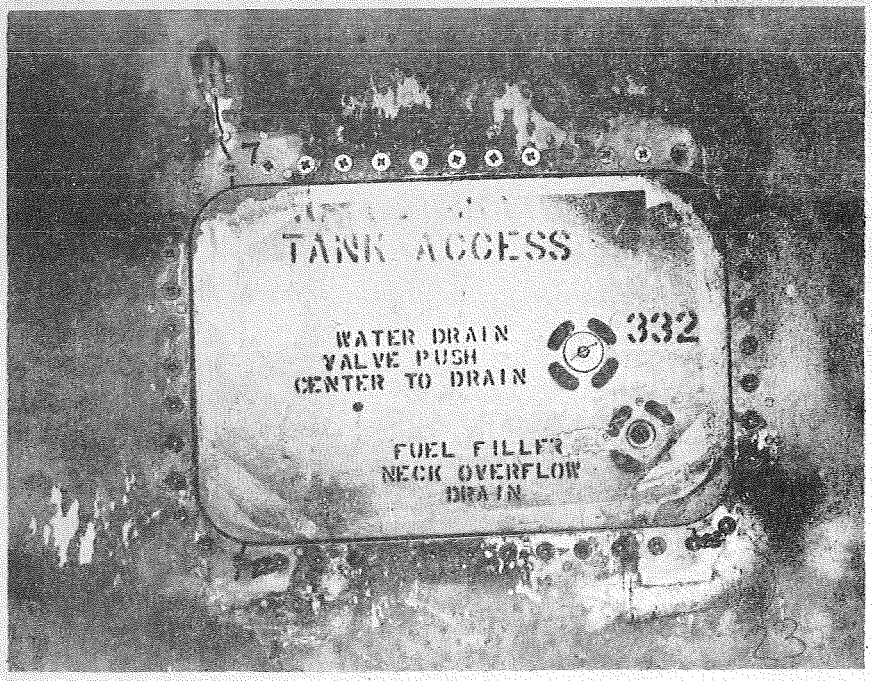


Figure 4-14 WS 120L, stringers 17 thru 19, lower surface fuel bag access door, subsequent to testing, showing items 7, 22 & 23.

RJ 5358-7

Figure 4-15 WS 112.8L, stringer 19 area, lower surface fuel bag access door, subsequent to testing, showing item 22.

RJ 5358-5

Figure 4-16 WS 112.8L, stringer 17 area, fwd inboard corner of fuel bag access door, subsequent to testing, showing item 7.

RJ 5358-6

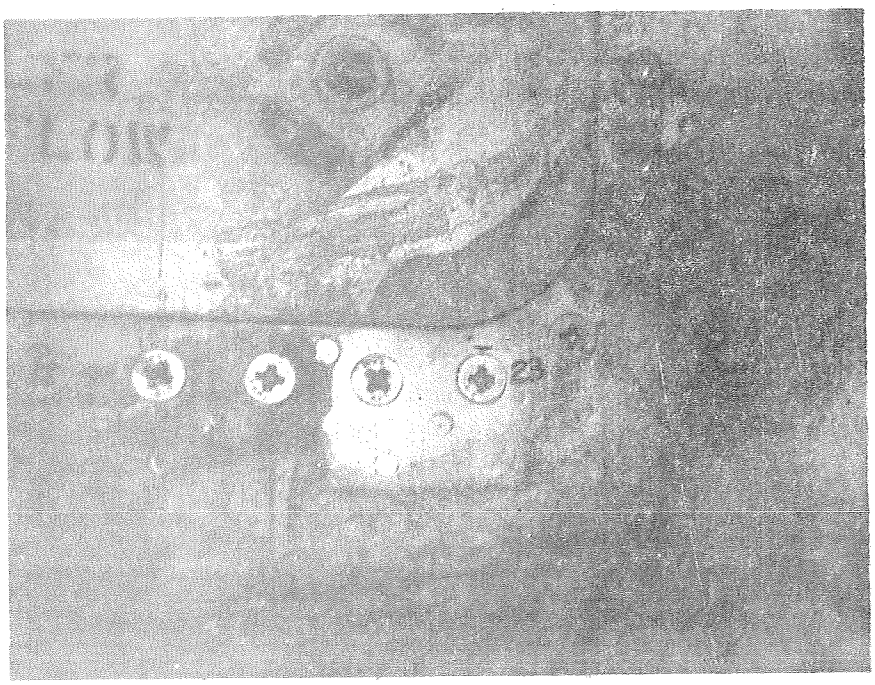


Figure 4-17 WS 128.2L, stringer 19 area, aft outboard of fuel bag access door, subsequent to testing, showing 23.

RJ 5281-5

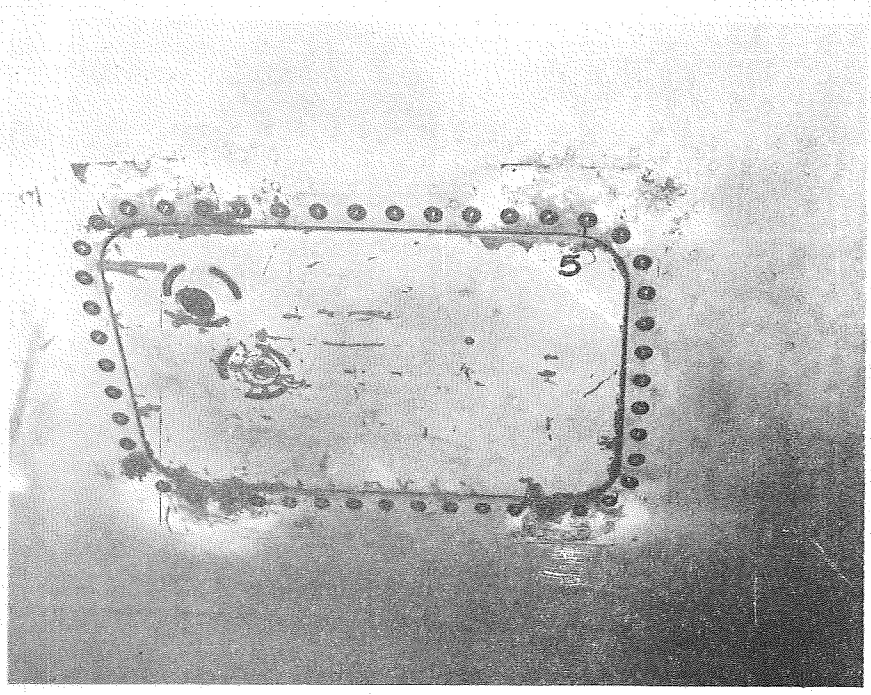


Figure 4-18 WS 120R lower surface fuel bag access door prior to testing showing item 5.

RJ 5366-5

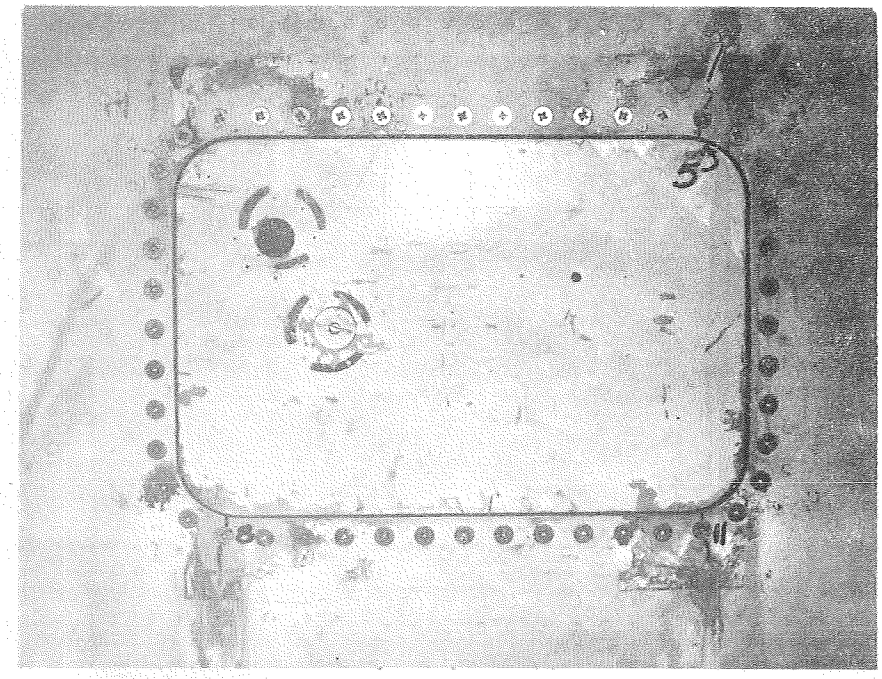
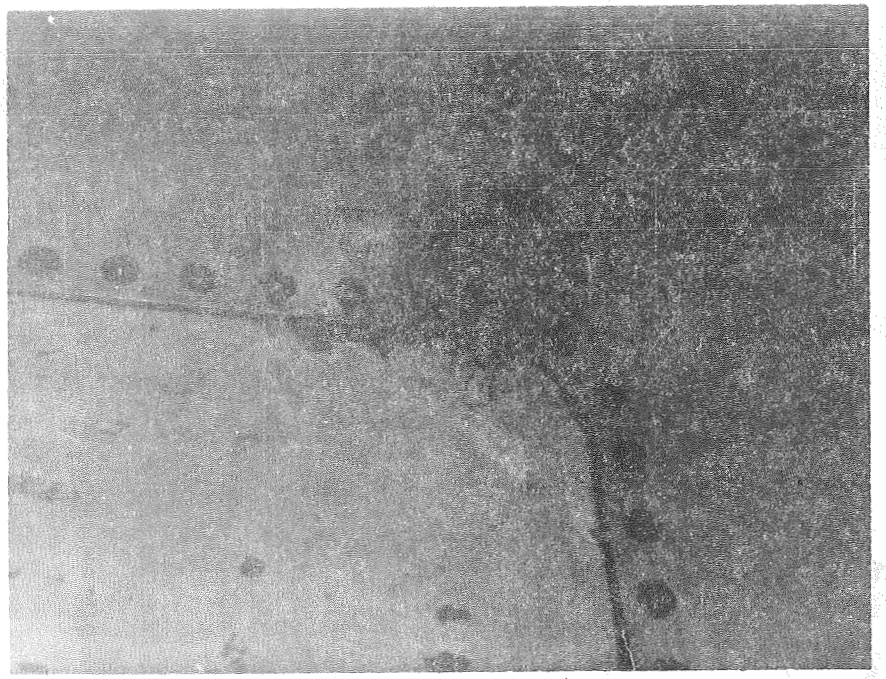
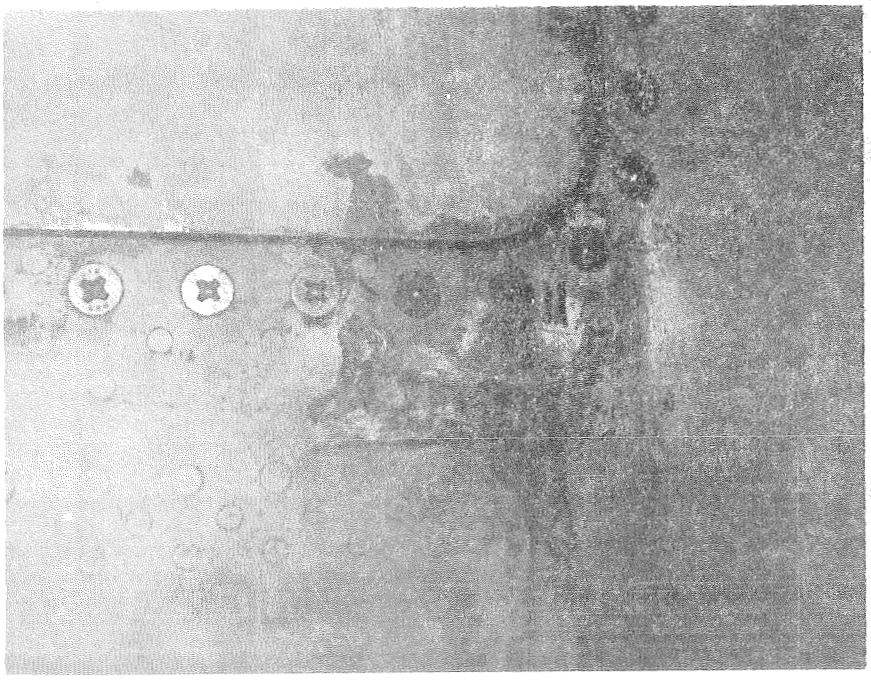


Figure 4-19 WS 120R lower surface fuel bag access door subsequent to testing, showing items 5, 8 & 11.



RJ 5358-13

Figure 4-20 WS 112.8L stringer 17 fwd inboard corner of fuel bag access door subsequent to testing showing item 5.



RJ 5358-12

Figure 4-21 WS 112.8L, stringer 19 aft inboard corner of fuel bag access door subsequent to testing showing item 11.

RJ 5358-11

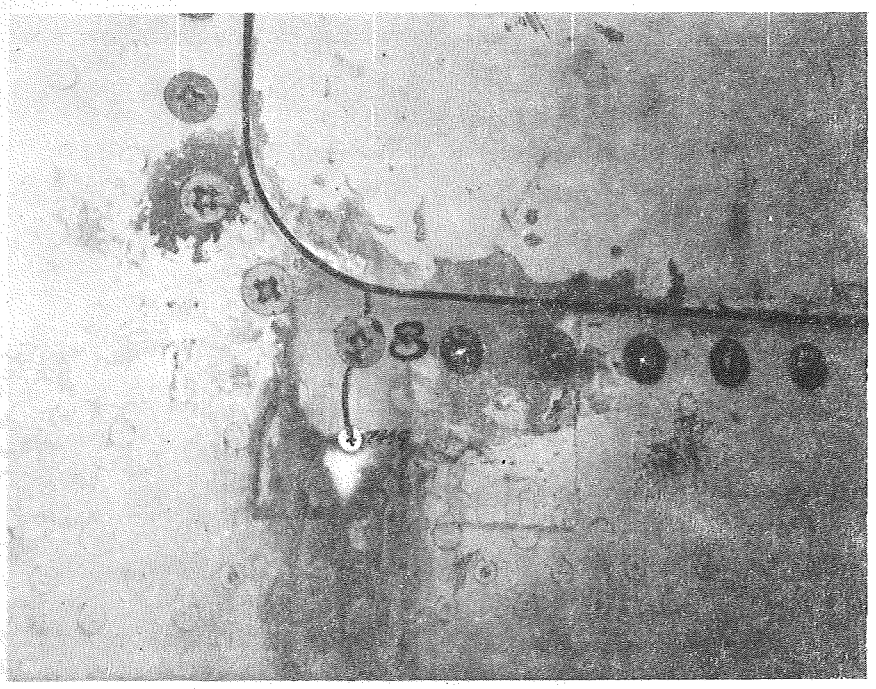


Figure 4-22 WS 128.2R stringer 19 aft outboard corner of lower surface fuel bag access door subsequent to testing, showing item 8.

PART 5

Specimen 5

CRACK PROPAGATION & RESIDUAL STRENGTH UPBENDING TEST RESULTS

TEST RESULTS

Crack 1, a 1.1-inch crack at W.S. 112.8R at the corner of the door cutout, was the only initial crack detected. (See Figure 5-8). This crack propagated to only 4.0 inches in the entire test, and was not associated with eventual specimen failure, as shown in Figure 5-9.

Initial cyclic load level was 50% Limit Load. Crack 10, a broken rainbow fitting node at Stringer #19 at W.S. 220L, was discovered at 7000 cycles (see Figure 5-11). At 7821 cycles, sonic reports were heard which led to discovery of three additional broken rainbow fitting nodes (cracks 16, 17, 18) at Stringers #12, #16, and #17 on the opposite wing at W.S. 220R. Cyclic load level was immediately reduced to 30% Limit Load, and cycling was resumed and continued until a total of 20,000 cycles had been applied. No further rainbow fitting node damage was observed during cyclic testing. Two cracks (5 and 6) of 5 inches or more length had developed at the end attachments in the reinforcing beams at W.S. 176R (see Figure 5-4) and W.S. 176L and 168L (see Figures 5-1 and 5-3). Other cracking is shown in Figures 5-2 and 5-7.

Residual strength test to destruction was accompanied by several sonic reports as load application proceeded in 10% increments from 50% to 80% Limit Load. Largest crack propagation occurred at W.S. 168L, with crack lengths of 8.8 inches (60%), 12.0 inches (70%), 24.4 inches (80%), and 26.1 inches (86.8%). At 80% four more right wing rainbow fitting nodes were broken at Stringers #13, #14, #15, and #18 (W.S. 220R). The rainbow fitting on the right wing failed completely at 86.8% Limit Load, as shown in Figure 5-10.

Table I-5 gives a summary of major fatigue damage accumulated by the specimen and Table II-5 gives a summary of minor fatigue damage.

TABLE I-5

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 5

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d)									
					0	500	2500	3000	4000	4500	5000	6000	6500	7000
1	The crack originated in the edge of a door attachment fastener hole	112.8R	0.8" aft of the stringer 17 ℄	5-8	1.1	1.6	2.2	2.6	3.2	———				3.2
			a	5-9	0.1	0.6(c)	1.2	1.6	2.2(c)	———				2.2(c)
			b		0.3(b)	0.3	0.3	0.3	0.3	———				0.3
4	Same as crack 1	129R	0.8" fwd of the stringer 19 ℄	5-9				0.3	1.4	1.5	1.7	2.0	2.2	2.2
			a					0.3(b)	0.3	0.3	0.3	0.3	0.3	0.3
			b						0.4	0.5	0.7	1.0	1.2(c)	1.2(c)
7	Same as crack 1	125R	0.5" fwd of the stringer 19 ℄	5-9								0.3	———	0.3
			a									0.3(b)	———	0.3
			b											
14	The crack originated in the edge of a fastener common to the W.S. 120 access door cutout reinforcement doubler and the skin	168L	At the stringer 17 ℄	5-1 5-6										
			a											
			b											
13	Same as crack 14 except fastener hole is also common to a stringer flange	168L	1.8" aft of the stringer 17 ℄	5-1 5-6										
			a											
			b											
12	Same as crack 13	168L	1.8" fwd of the stringer 18 ℄	5-1 5-6										
			a											
			b											
6	The crack originated in the edges of fastener holes common to the W.S. 120 door cutout reinforcement beam, the skin, and a 'slingshot' repair doubler	176L	0.5" fwd & aft of the stringer 16 ℄	5-4 5-6						1.6	———			1.6
			a							1.6(b)	———			1.6
			d											
9	The crack originated in the edges of fastener holes common to the W.S. 120 door cutout reinforcement beam and the skin	175L	0.5" fwd & aft of the stringer 21 ℄	5-5 5-6										0.9
			a											–
			b											0.2
			c											0.1
			d											0.2
5	Same as crack 6	176R	0.5" fwd & aft of the stringer 16 ℄	5-3						1.5	3.6	4.6	4.9	5.4
			a							1.5	1.5	1.6(c)	1.6	1.6
			d								0.5	1.1	1.5	1.9

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-5 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 5

LOWER SURFACE SKIN (s)

Crack No.	Phase 1		Upbending, Phase 2 Cyclic Test, R= 0.1 Max. Load = 30% Limit Load									Upbending Residual Strength Test		
	Cumulative Number of Cycles (d)											After Applying 60% Limit Load	After Applying 80% Limit Load	After Applying 86.8% Limit Load
	7500	8000	14,500	15,000	15,500	16,000	17,500	18,000	18,500	19,500	20,000			
1	3.5	3.7				3.7	4.0				4.0		4.0	5.5
a	2.5	2.7				2.7	3.0				3.0		3.0	4.5
b	0.3	0.3				0.3	0.3				0.3		0.3	0.3
4	2.2													2.2
a	0.3													0.3
b	1.2													1.2
7	0.3	1.3												1.3
a	0.3	0.3												0.3
b	-	0.3												0.3
14		0.1					0.1	0.2	0.2					
a														
b		0.1					0.1	0.2	0.2					
13		0.7	0.8	0.8					0.8					
a		0.2	0.2	0.2					0.2					
b		0.1	0.2	0.3					0.3					
12	0.9	0.9	0.9	0.9	1.1	3.2	3.7	3.9	4.2	5.9	6.6	8.8	24.4	26.1
a	0.4	0.4	0.4	0.4	0.6	2.4(c)	2.6	2.7	2.9	3.5	4.0	5.1	8.8	9.9
b	0.3	0.3	0.4	0.4	0.5	0.5	0.7	0.9	1.0	2.0	2.2	3.3	14.7	15.3
6	1.6	4.9	4.9	5.0	5.0	5.0	5.2	5.3	5.3	5.3	5.6			5.6
a	1.6	1.6	1.6	1.6	1.6	1.6	1.6	1.6	1.6	1.6	1.6			1.6
b		1.5	1.5	1.6	1.6	1.6	1.8	1.9	1.9	2.0	2.2			2.2
9	0.9	0.9	1.2	1.2	1.3	1.3	1.9	1.9	2.0	2.0	2.1		2.1	2.9
a	-	-		-			0.1	0.1	0.1	0.1	0.2		0.2	0.8
b	0.2	0.2	0.4	0.4	0.4	0.4								
c	0.1	0.1	0.4	0.4	0.4	0.4								
d	0.2	0.2	0.3	0.3	0.3	0.3	0.4	0.4	0.5	0.5	0.5		0.5	0.7
5	6.2	6.8									6.8		6.8	9.6
a	1.6	1.6									1.6		1.6	1.6
b	2.6	3.2									3.2		3.2	6.7

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-5 (Cont.)
SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 5
LOWER SURFACE SKIN (a)

No.	Location	Fig.			Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load
10	The crack originated in the edge of a bolt access hole in wing joint fitting	220L	0.3" aft of the stringer 18 ℄	5-11	(f)
19	Same as crack 10		0.3" aft of the stringer 12 ℄		
20			0.3" aft of the stringer 13 ℄		
21			0.3" aft of the stringer 14 ℄		
22			0.3" aft of the stringer 15 ℄		
23			0.3" aft of the stringer 16 ℄		
24			0.3" fwd of the stringer 17 ℄		
25	Same as crack 10	220L	0.3" aft of the stringer 17 ℄	5-11	
18	Same as crack 10	220R	0.3" aft of the stringer 12 ℄	5-10	
26			0.3" aft of the stringer 13 ℄		
27			0.3" aft of the stringer 14 ℄		
28			0.3" aft of the stringer 15 ℄		
16			0.3" aft of the stringer 16 ℄		
17			0.3" aft of the stringer 17 ℄		
29	Same as crack 10	220R	0.3" aft of the stringer 18 ℄	5-10	

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-5 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 5

LOWER SURFACE SKIN (a)

Crack No.	Phase 1		Upbending, Phase 2 Cyclic Test, R = 0.1 Max. Load = 30% Limit Load								Upbending Residual Strength Test		
	Cumulative Number of Cycles (d)										After Applying 60% Limit Load	After Applying 80% Limit Load	After Applying 86.8% Limit Load
	7500	7821	14,500	15,000	15,500	16,000	17,500	18,000	18,500	19,500	20,000			
10	(f)	———											(f)
19													(f)
20													(f)
21													(f)
22													(f)
23													(f)
24													(f)
25													(f)
18		(f)	———									(f)	
26												(f)	Complete Chordwise Failure
27												(f)	
28												(f)	
16		(f)	———									(f)	
17		(f)	———									(f)	
29												(f)	

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE II-5

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 5

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered (Incnes)	(d) Length After 20,000 Cycles	Length After Failure at 86.8%
2	The crack originated in the steel heat shield at the edge of a fastener hole common to the skin and the heat shield	196L	0.7" aft of the stringer 22 ℄ a b	5-2 5-6	1200	0.4 0.4	0.9 0.4 0.3	0.9 0.4 0.3
3	The crack originated in a drain hole in the skin	179L	2.8" aft of the stringer 19 ℄ a b	5-6	2750	0.1 0.1	0.5 0.1 0.1(c)	0.5 0.1 0.1
8	The crack originated in the edge of a door attachment fastener	112.8L	0.8" fwd of the stringer 20 ℄ a	5-7	6568	0.3 0.3(b)	0.3 0.3	0.3 0.3
11	Same as crack 8	128.2R	0.8" aft of the stringer 17 ℄ b	5-9	7231	0.3 0.3(b)	0.3 0.3	0.3 0.3

NOTE: Footnotes found in Table III of the General Section, Part C.

GA FORM 423A-1

RJ 5493-3

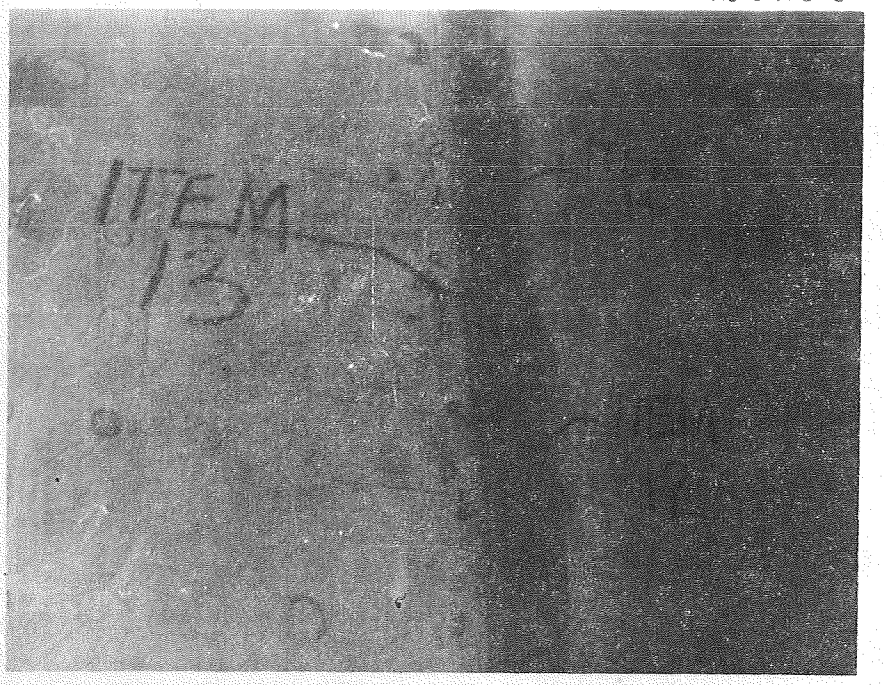


Figure 5-1 WS 168L, stringers 17-19 following crack propagation test, showing items 12, 13 & 14.

RJ 5493-2

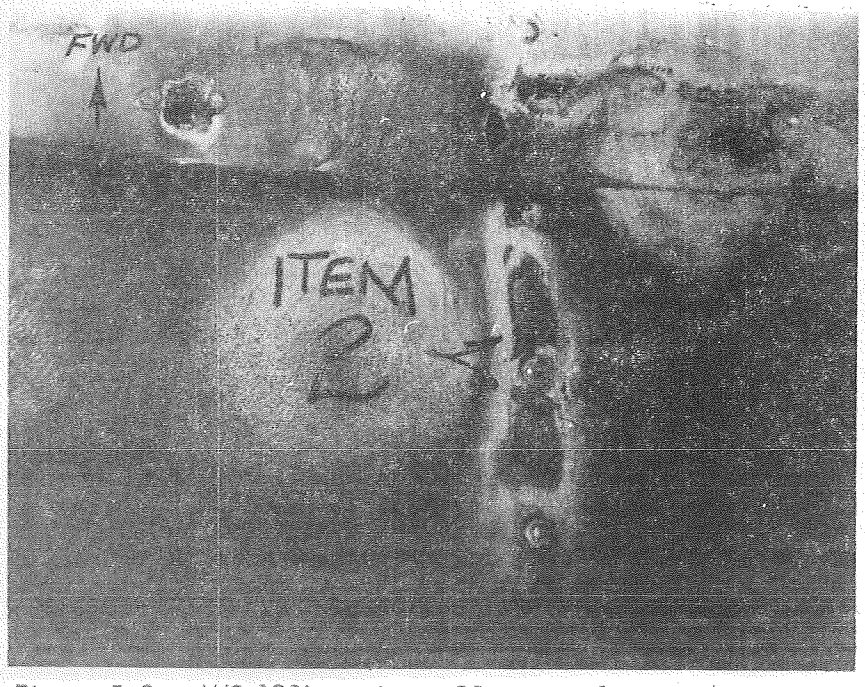


Figure 5-2 WS 190L, stringer 22 area, after crack propa-

RJ 5493-8

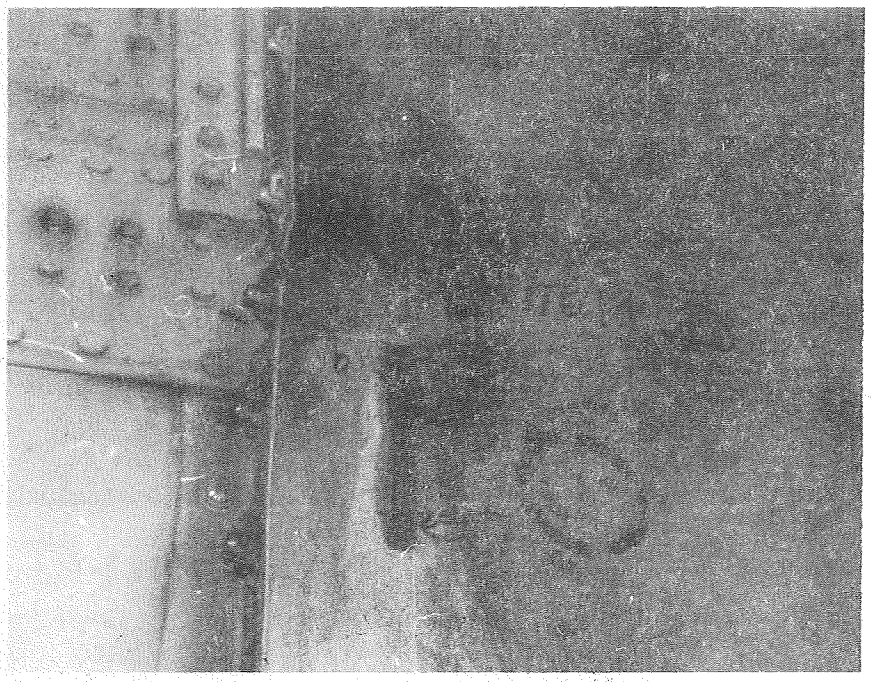


Figure 5-3 WS 176L, stringers 15 & 16 area after crack propagation testing showing item 5.

RJ 5493-4

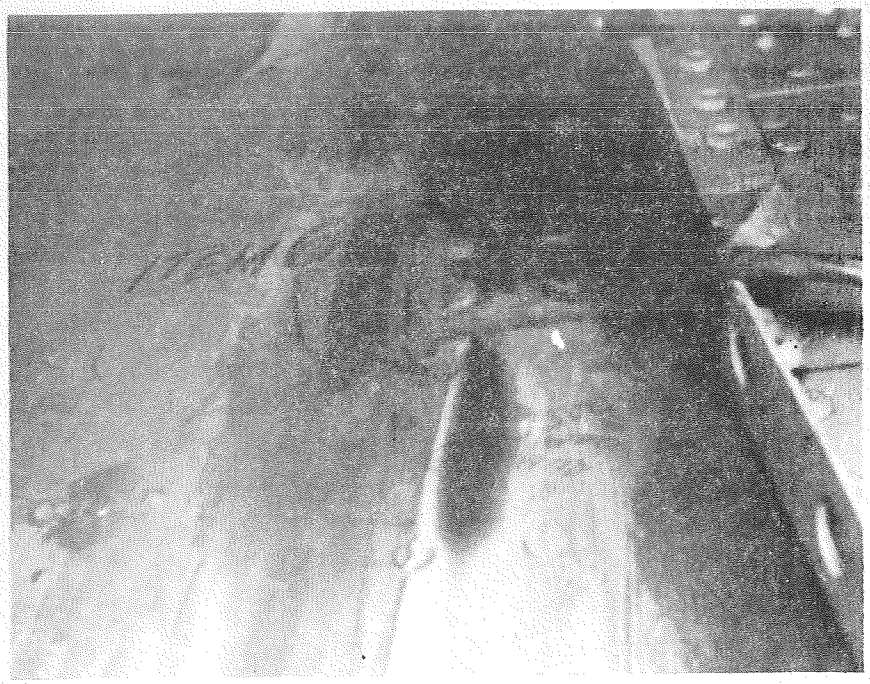


Figure 5-4 WS 176R, stringers 15 & 16 area. After crack propagation testing, showing item 6.

RJ 5493-1

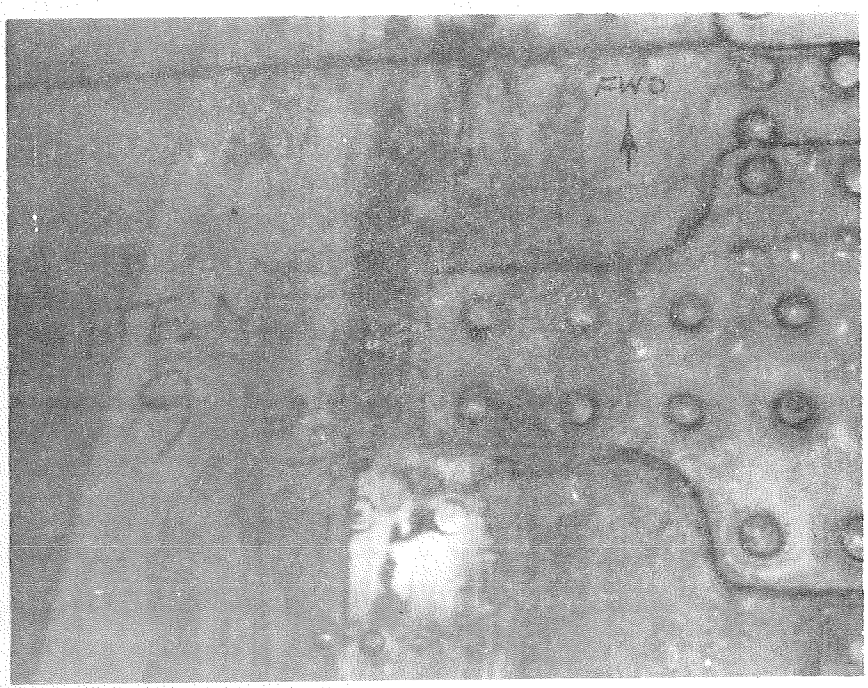


Figure 5-5 WS 176R, stringer 21 area, after crack propagation testing view showing item 9.

RK 0026-11

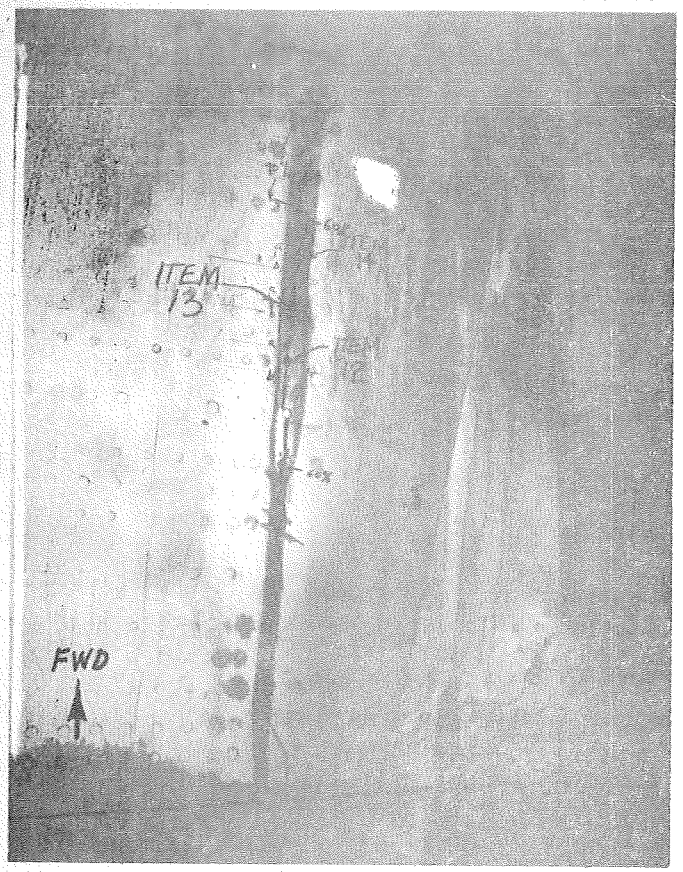
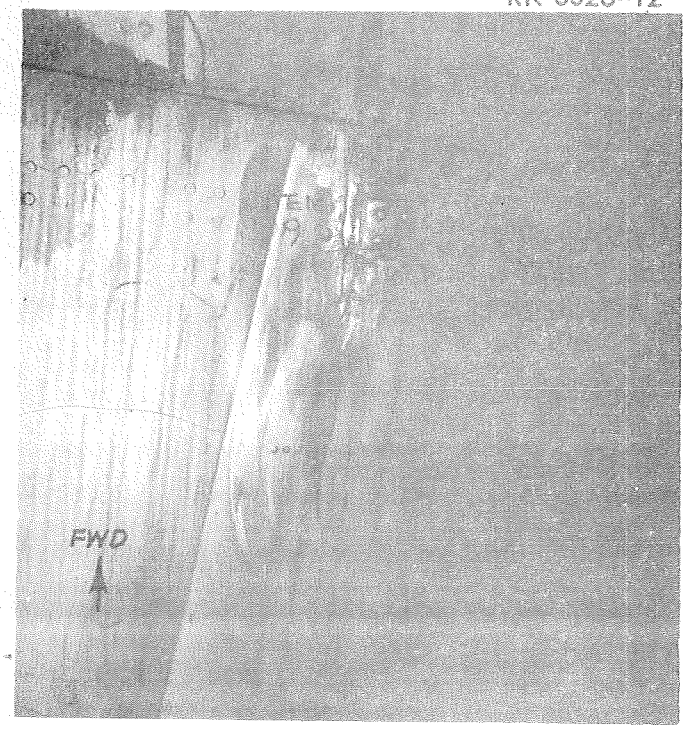


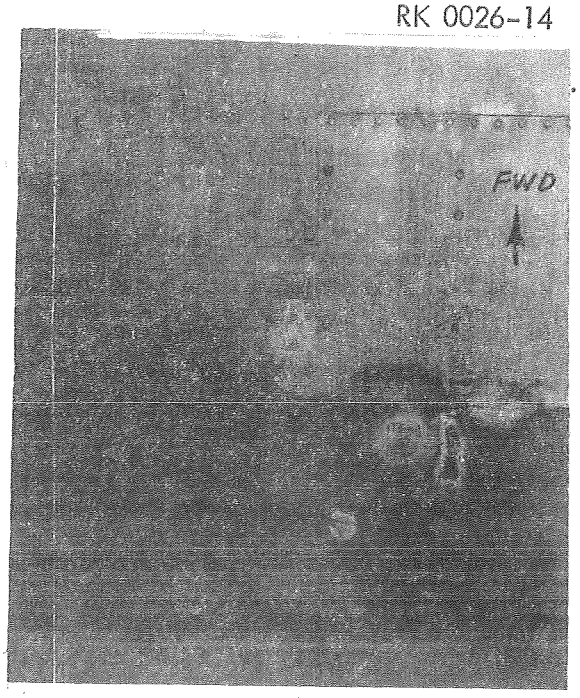
Figure 5-6

WS 176L/ 168L area after failure, showing items 6, 12, 13, 14, 3 & 2.

RK 0026-12



RK 0026-14



RJ 5493-6

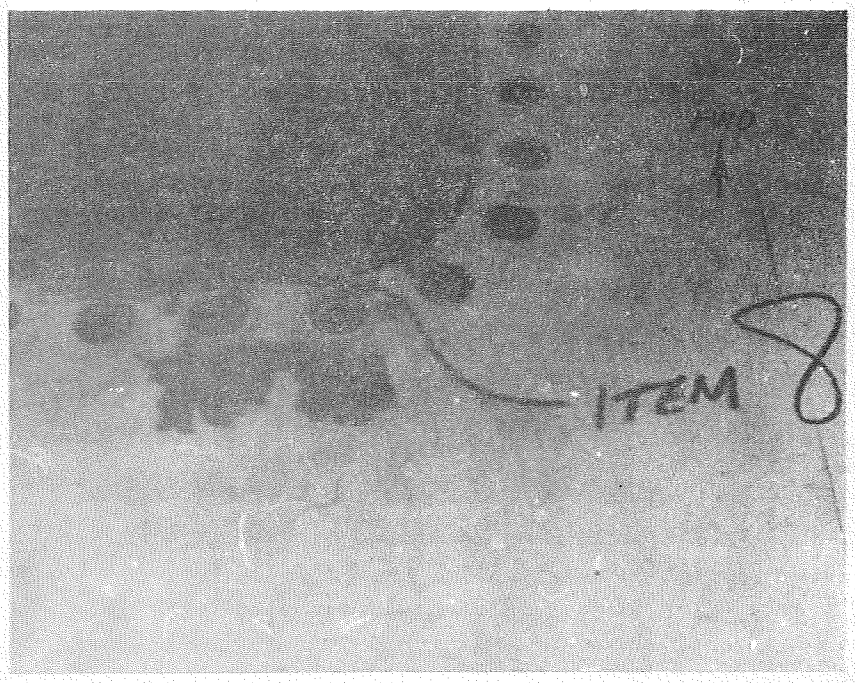


Figure 5-7 Damage to WS 120L door subsequent to testing. Showing item 8.

RJ 5413-2

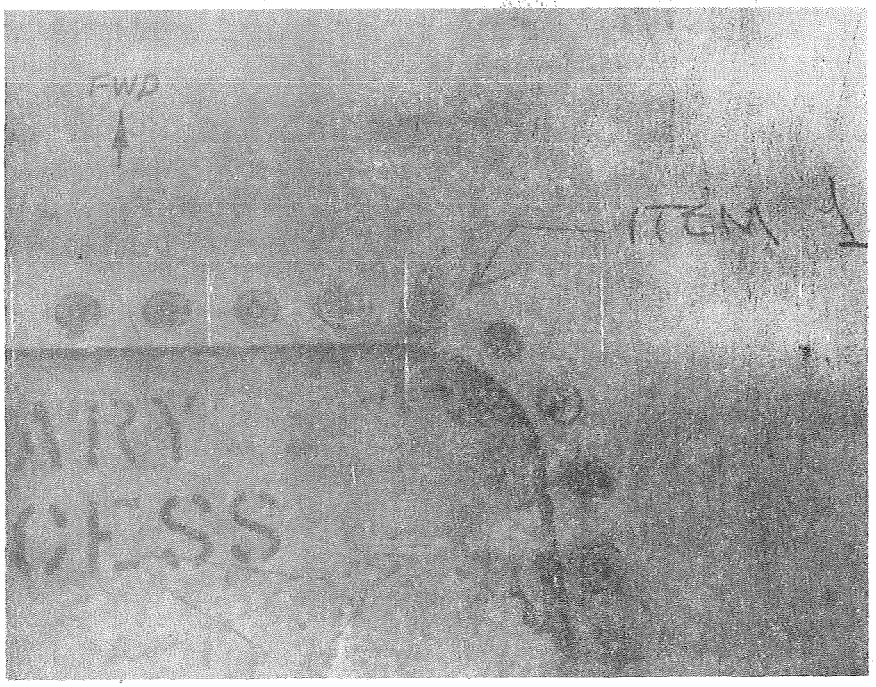


Figure 5-8 Damage to WS 120R lower surface access door prior to testing, showing item 1.

RK 0026-8

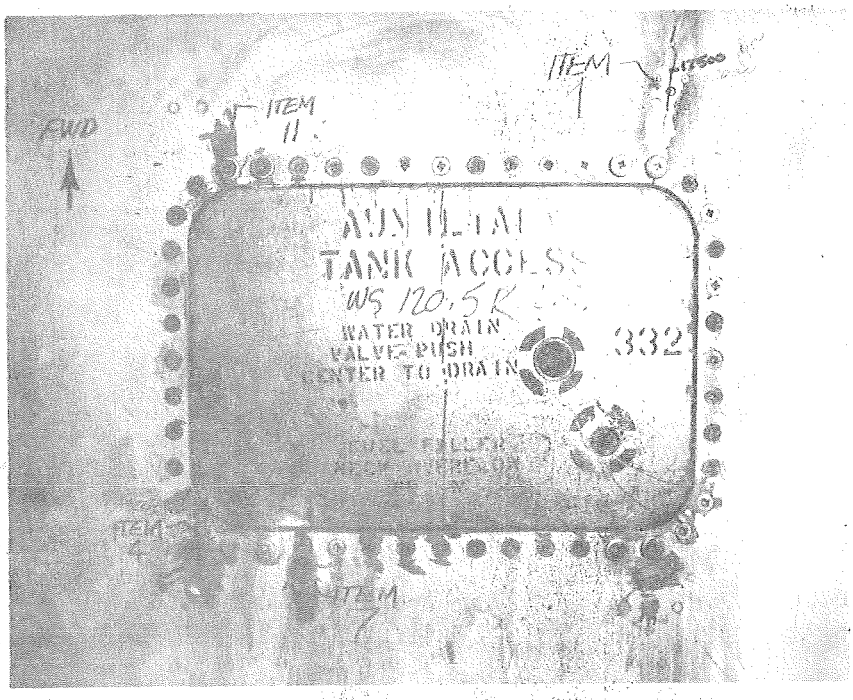
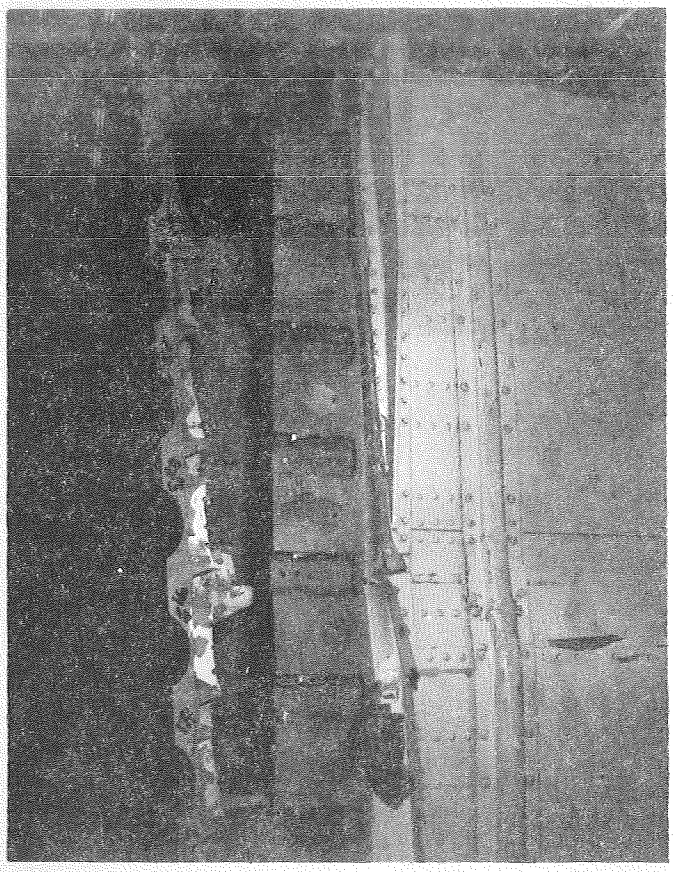


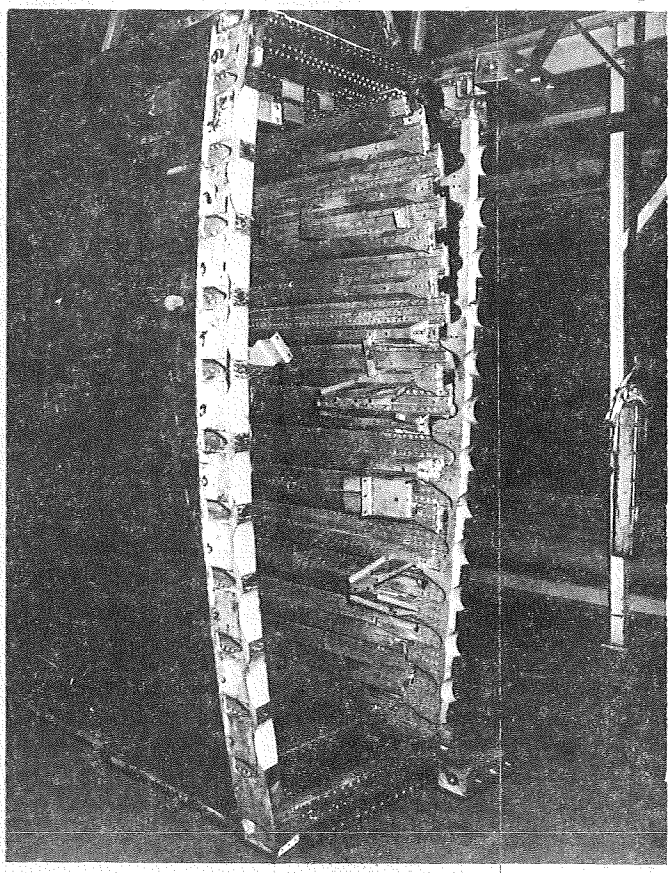
Figure 5-9 Damage to WS 120R lower surface access door subsequent to testing showing items 1, 4, 7 & 11.

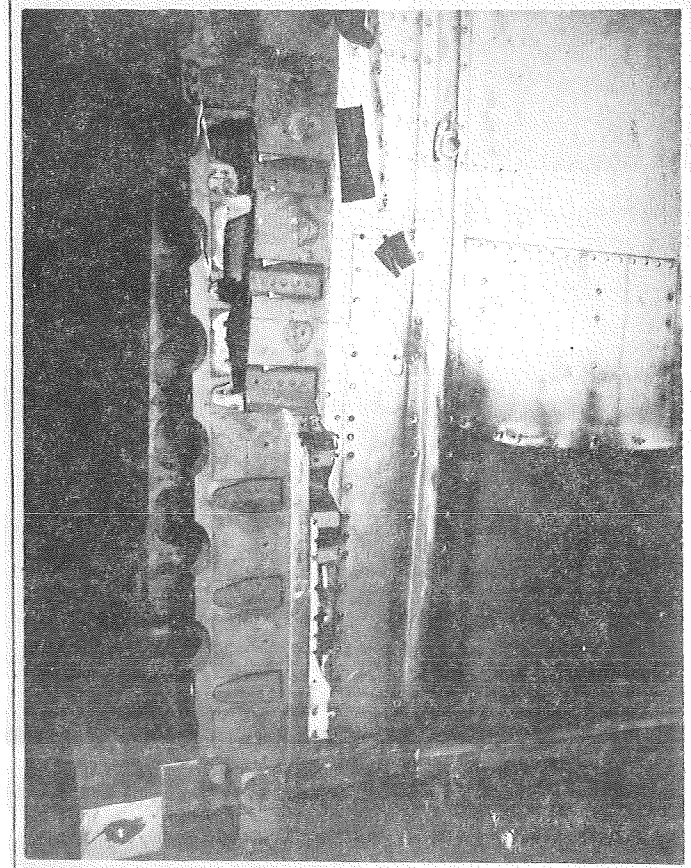


RK0026-1

Figure 5-10
WS 220R rainbow fitting after failure.

RK0026-6





RK0026-2

RK 0026-13

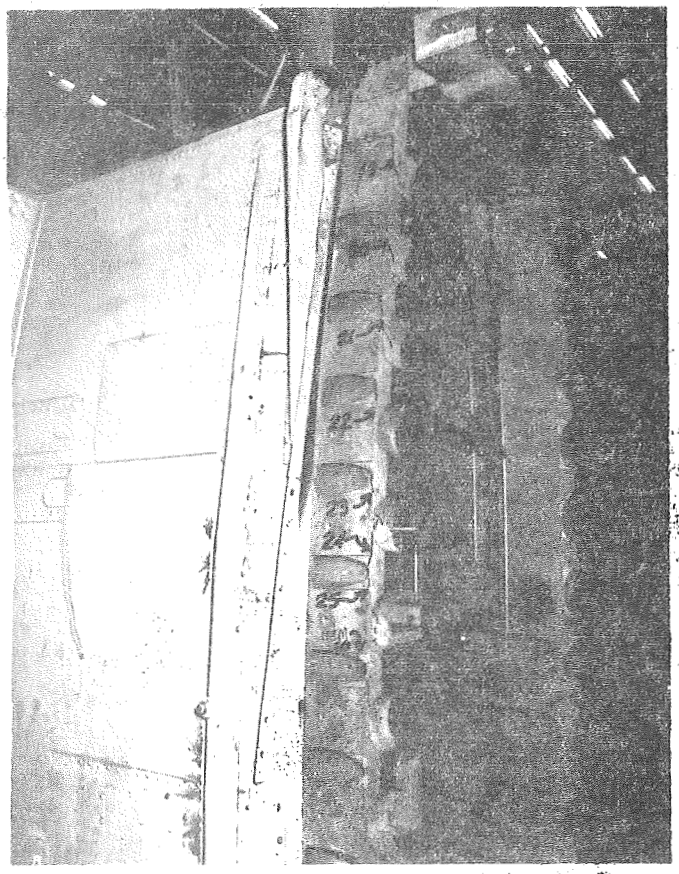


Figure 5-11 WS 220L rainbow fitting after failure.

RK 0026-7

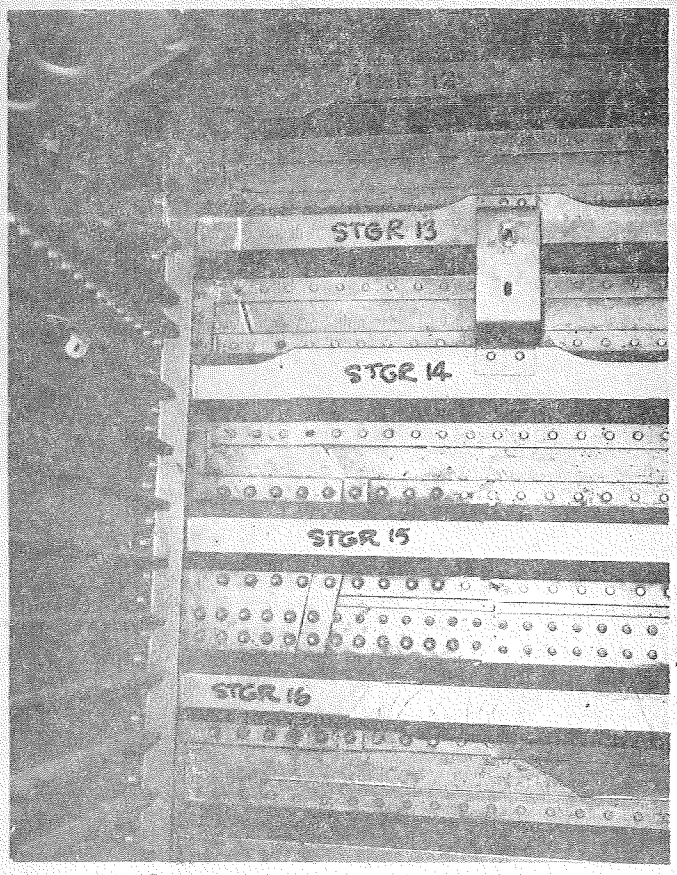


Figure 5-12 Internal view showing secondary damage to stringers & skin at WS176-182R, stringer 12 thru 16 area after failure.

Part 6

Specimen 6

RESIDUAL STRENGTH
DOWNBENDING TEST RESULTS

TEST RESULTS

Figures 6-1 and 6-2 show the largest initial cracks, numbers 1 and 2, which occurred at W.S. 34.5L along the end row of fasteners attaching the skin to the reinforcing doubler around the cutout. Propagation of these cracks during static testing is shown in Figure 6-3. Other significant cracking is shown in Figures 6-4, 6-5, and 6-6. The crack propagation in the aft direction from the fuel filler access hole at W.S. 120.5L (Figure 6-5) was arrested by the panel splice. This was the most crack propagation at the fuel filler access hole observed in any of the 14 tests, even though initial cracking at this location existed on several of the test specimens.

Primary failure occurred at 96% Limit Load (Condition D-1, Figure 11) due to stripping of all barrel nut threads in the tension surface connections at the W.S. 220 joint. Massive secondary failures occurred in the nearby box surface and beam structure (See Figure 6-7). Failure at this location and load level occurred partly due to the static overload associated with the use of Condition D-1 which should be (and subsequently was) used only as a cyclic test loading which was not applied at higher than 50% Limit Load. Another potential contributing factor was the possibility that the end bolts which introduce the tension surface component of the bending moment had "bottomed out" against the test fixture and therefore were inadvertently over-torqued during installation. On all subsequent test specimens, washers were installed under the bolt heads to preclude any possible "bottoming out" of the bolts. No further problems occurred with the W.S. 220 joint connection on any of the remaining test specimens.

Crack history for this specimen appears in Table I-6.

TABLE I-6

SUMMARY OF CRACK LENGTHS, SPECIMEN 6

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Initial Crack Length, In.	Downbending 1 Residual Strength Test (d)		
						After Applying 77.8% Limit Load	After Applying 90% Limit Load	After Failure at 96.3% Limit Load
1	The crack originated in the edge of a last fastener hole common to the internal reinforcing doubler of the access door cutout and the skin	34.5L	The stringer 5 ℄ d	6-1	4.4 1.7		7.0	7.0
2	Same as crack 1	34.5L	0.8" aft of the stringer 4 ℄	6-2 6-3	0.3		7.1	7.1
3	Same as crack 1	32.2L	3.7" aft of the stringer 4 ℄			0.9	1.2	1.2
4	The crack originated under the circular external repair doubler	120.5R	2.5" fwd of the stringer 6 ℄	6-6		0.3(c)	———	0.3
6	Same as crack 4	120.5L	2.5" fwd of the stringer 6 ℄	6-5		0.5	4.1(b)	4.1
7	Same as crack 4	120R	2.5" aft of the stringer 4 ℄	6-6			0.3	0.3
8	Same as crack 1	37.5R	Same as 1	6-4			10.7(b)	10.7
9	Barrel nuts attaching the specimen to the test frame were stripped resulting in specimen failure at the wing joint	220R	Rainbow fitting	6-7				Complete Chordwise Failure
5	Same as crack 4	120.5L	2.5" aft of the stringer 4 ℄	6-5		0.3(c)	———	0.3

NOTE: Footnotes found in Table III of the General Section, Part G.

RK 0026-15

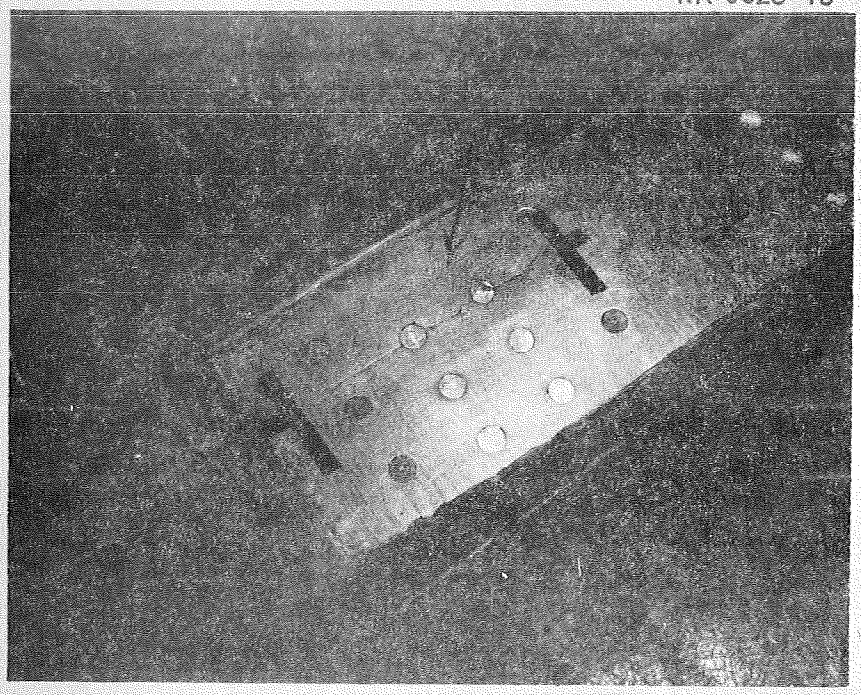


Figure 6-1 Crack 1 passes through last fastener holes common to the access door cutout reinforcing doubler.

RK 0026-2

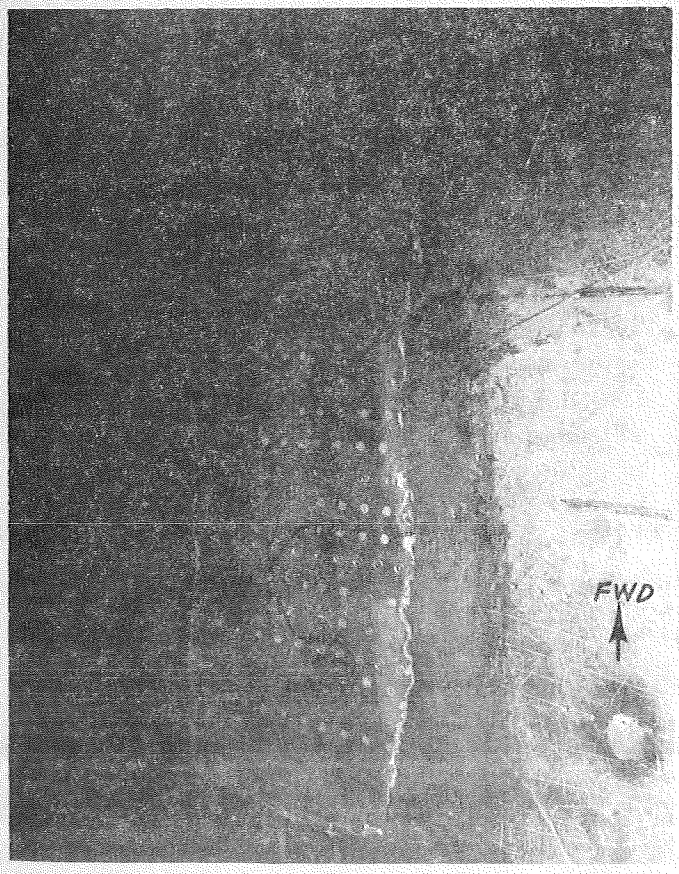
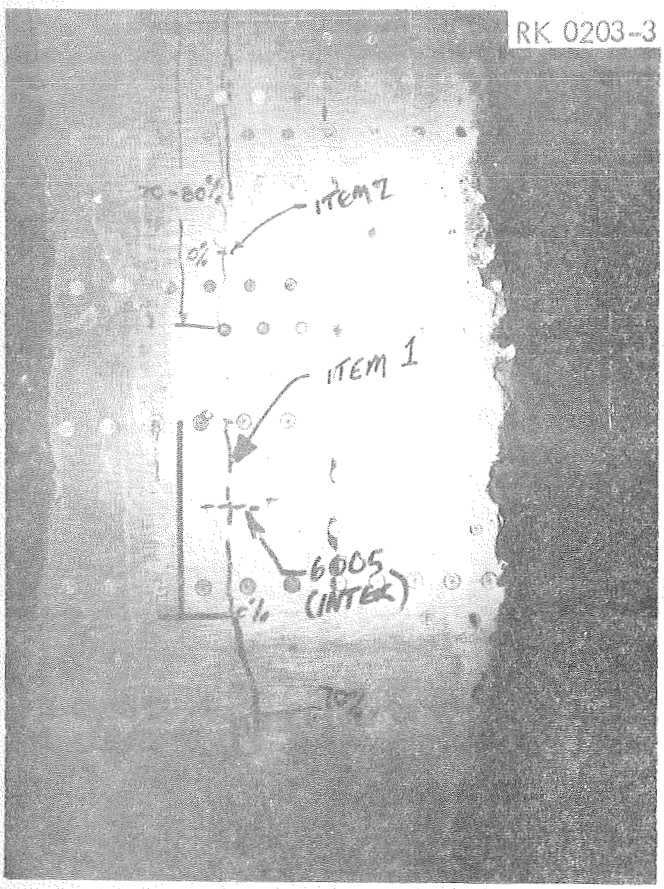
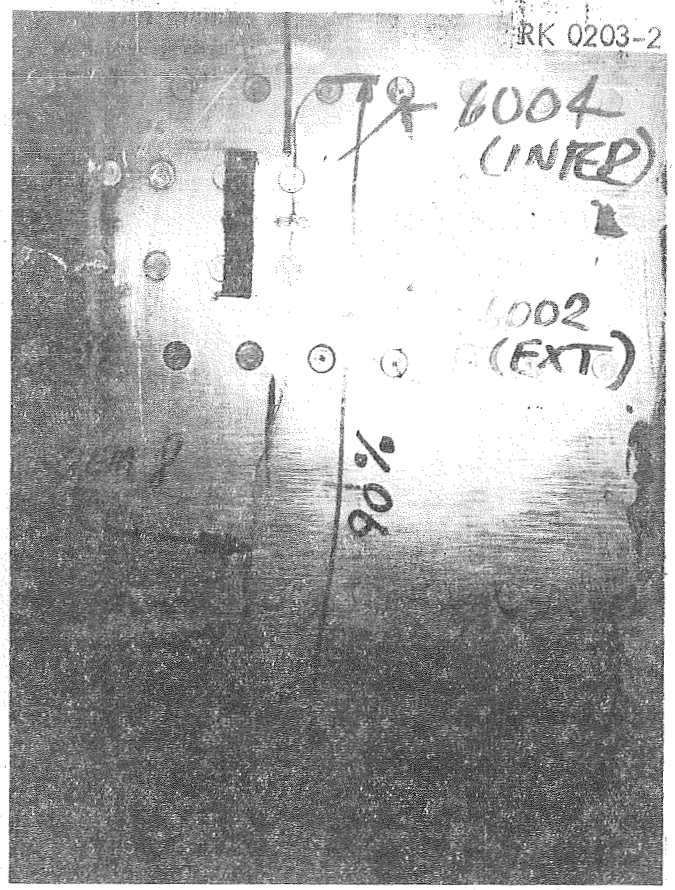


Figure 6-2
Locations of cracks 1 and 2 (WS 34.5L)





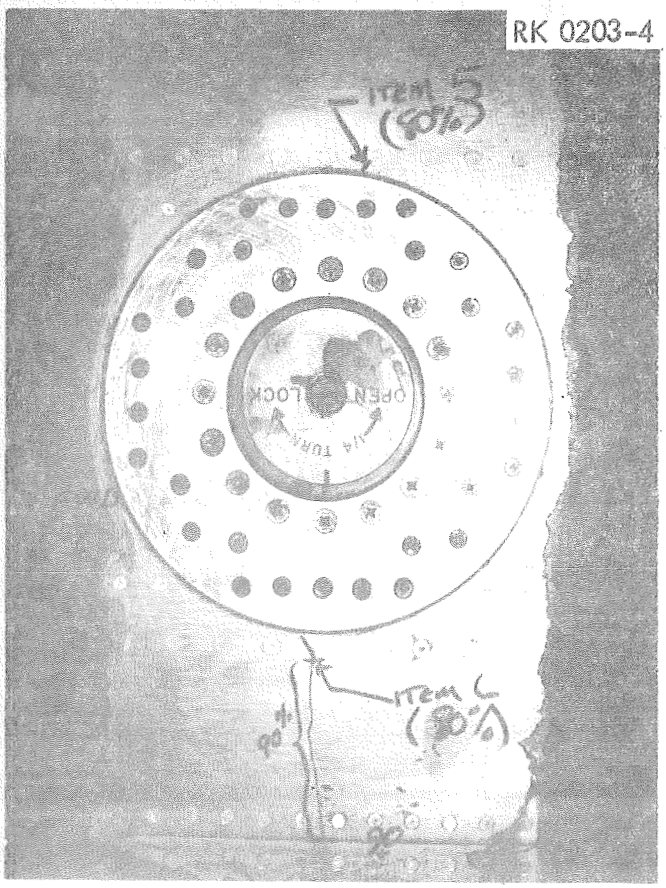


Figure 6-3 (Upper Left)
Crack 1 extended aft to fastener hole common to the stringer 6 leg after applying 70% of limit load. After 80% crack 2 extended fwd and aft as shown.

Figure 6-4 (Upper Right)
Crack 8 discovered after 90% of limit load was applied. Crack extended through last fastener attaching reinforcing doubler to skin at WS 37.5R.

Figure 6-5 (Left)
Cracks 5 and 6 are located at WS 120.5L.

RK 0203-1

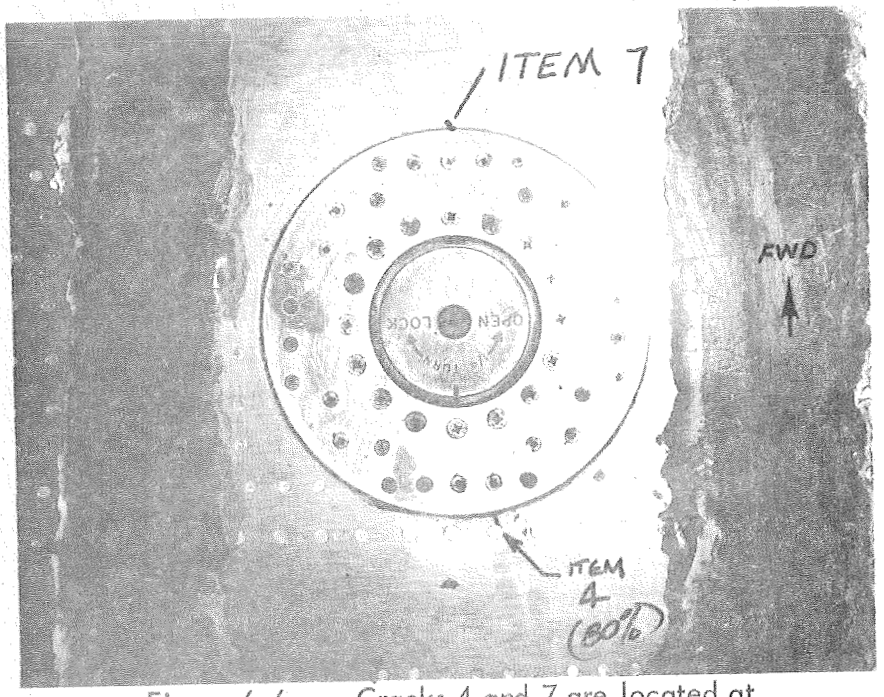


Figure 6-6 Cracks 4 and 7 are located at WS 120.5R.

RK 0203-5

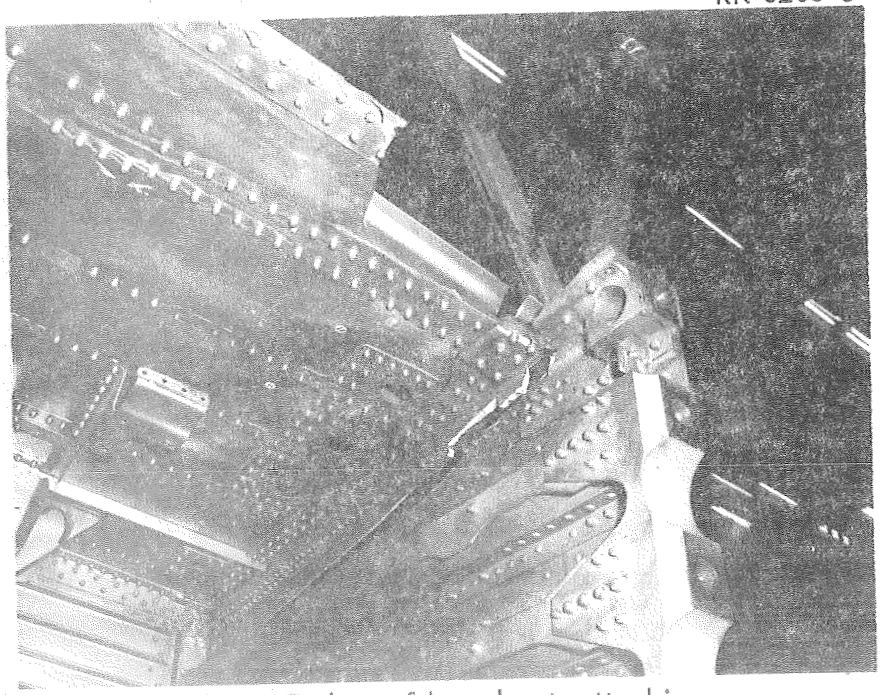


Figure 6-7 Failure of barrel nuts attaching specimen to fixture resulted in damage shown.

PART 7

Specimen 7

RESIDUAL STRENGTH
DOWNBENDING TEST RESULTS

TEST RESULTS

This was the only C-130B model tested (all others were C-130E models). Extensive corrosion degradation during service had occurred, particularly in the upper surface panels adjoining the front and rear beam caps, but this had no evident relationship to the initiation or propagation of fatigue cracking. Principal initial cracking is shown in Figures 7-1 and 7-4.

From the crack history record of Table I-7, it may be observed that the longest initial crack (2.3 inches) and the maximum initial total crack-length (crack 2 + crack 3 = 3 inches) existed at WS 105L. While these cracks grew from 3.0 inches to 9.6 inches at the end of the test, failure occurred at W.S. 135 (R). Largest initial crack at W.S. 135 (R) was 1.4 inches, and total initial crack length at this station was 2.7 inches. Thus, the location of maximum initial crack length was not where the failure occurred.

At 97.7 percent of limit load total chordwise failure of the specimen occurred along the reinforcing doubler runout area located at W.S. 135.3R. Figures 7-6 and 7-7 show details of the fracture surface. New damage inflicted at locations away from the primary failure is shown in Figures 7-8 (W.S. 135L) and 7-9 (120.5R).

A summary of damage is given in Table I-7.

TABLE I-7
SUMMARY OF CRACK LENGTHS, SPECIMEN 7
UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Downbending 2 Residual Strength Test (d)		
					Initial Crack Length, In.	After Applying 72.3% Limit Load	After Failure at 97.7% Limit Load
1	The crack originated in the aft attachment hole of the fuel filler cutout	120.5L	1.8" aft of the stringer 5 ℄	7-1 7-2	0.3(c)		1.6(c)
2	The crack originated in the edge of the last fastener hole common to the internal reinforcing doubler of the fuel filler cutout	105.5L	2.7" aft of the stringer 4 ℄	7-1 7-3	0.7		9.6(c)
3	Same as crack 2	105.5L	2.7" fwd of the stringer 6 ℄	7-1 7-3	2.3	2.7	
4	Same as crack 2	105.5R	2.7" aft of the stringer 4 ℄	7-4 7-5	0.2		0.2
5	Same as crack 2	105.5R	0.8" aft of the stringer 5 ℄	7-4 7-5	0.5		0.5
6	Same as crack 2	105.5R	2.7" fwd of the stringer 6 ℄	7-4 7-5	1.4(e)		1.9(c)
7	Same as crack 2	135.3R	2.7" aft of the stringer 4 ℄	7-4 7-6	0.7		Complete Chordwise Failure
8	Same as crack 2	135.3R	0.8" aft of the stringer 5 ℄	7-4 7-6	0.5		
9	Same as crack 2	135.3R	2.7" fwd of the stringer 6 ℄	7-4 7-6	1.4		
10	Same as crack 2	135.3R	2.7" fwd of the stringer 5 ℄	7-6 7-7	0.1		
11	Same as crack 2	135.3L	1.8" aft of the stringer 5 ℄	7-8			0.7
12	Same as crack 2	135.3L	0.8" aft of the stringer 5 ℄	7-8			0.8
13	Same as crack 1	120.5R	1.8" fwd of the stringer 5 ℄	7-9	0.3(c)		0.3
14	Same as crack 1	120.5R	1.8" aft of the stringer 5 ℄	7-9			1.7(c)

NOTE: Footnotes found in Table III of the General Section, Part G.

RK 0203-8

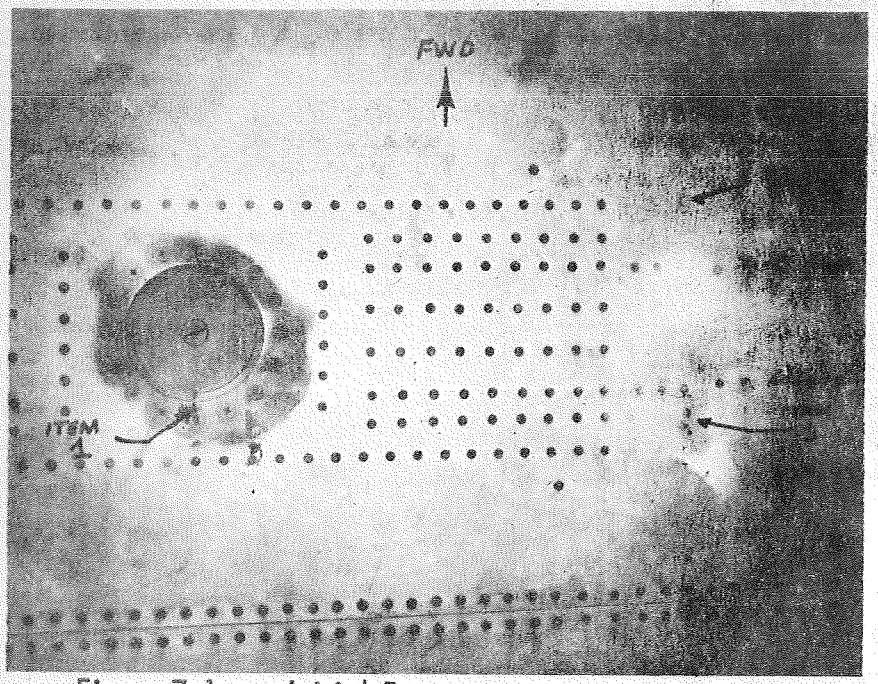


Figure 7-1 Initial Damage
Crack 1 is located at the fuel filler cutout (WS 120.5L). Cracks 2 and 3 are located in the doubler runout area (WS 105.5L).

RK 0259-8

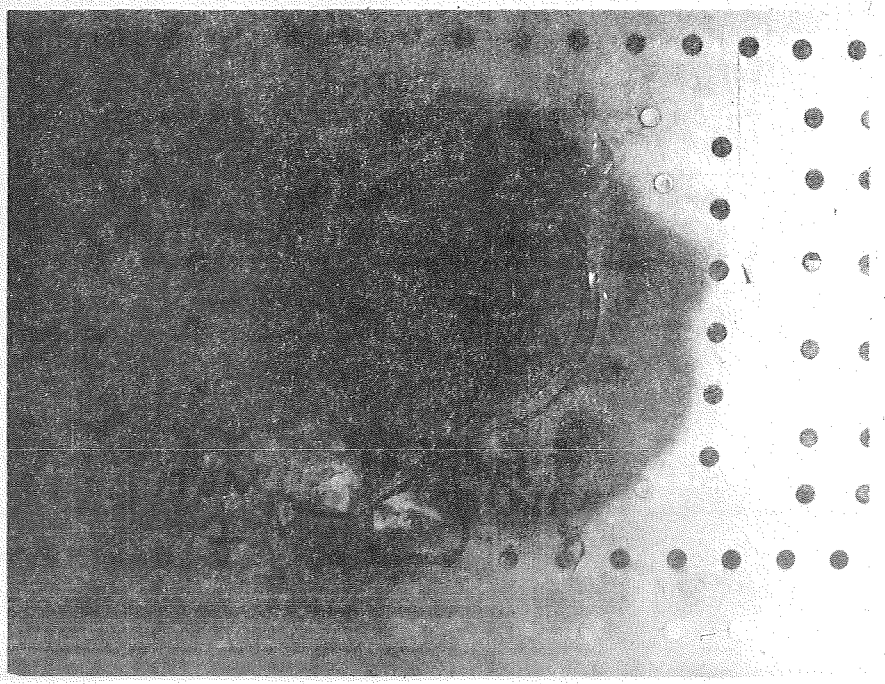


Figure 7-2 Propagation
Crack 1 extended aft into fastener hole at 97.7% of limit load.

RK 0259-5

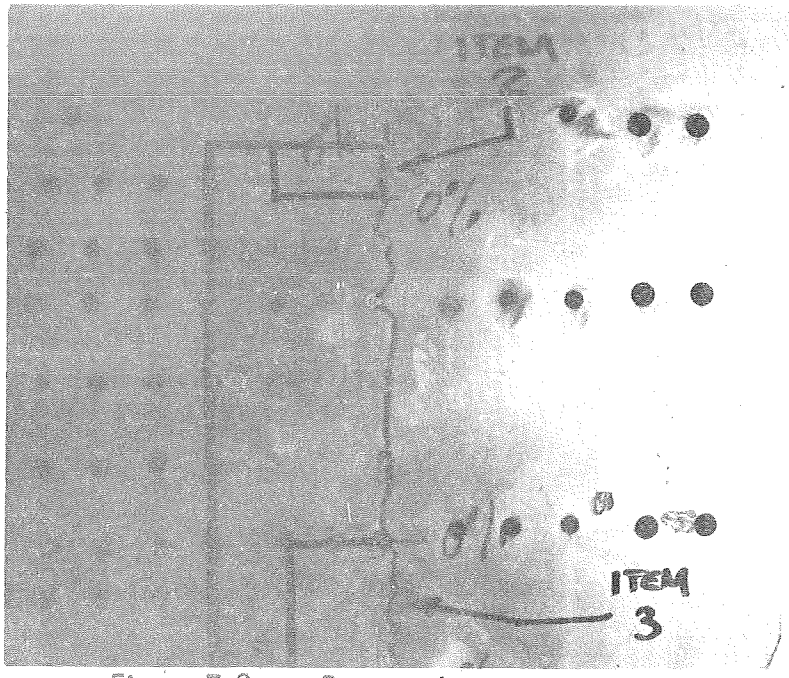


Figure 7-3 Propagation
Cracks 2 and 3 join at 97.7% of limit load.

RK0203-7

RK 0203-6

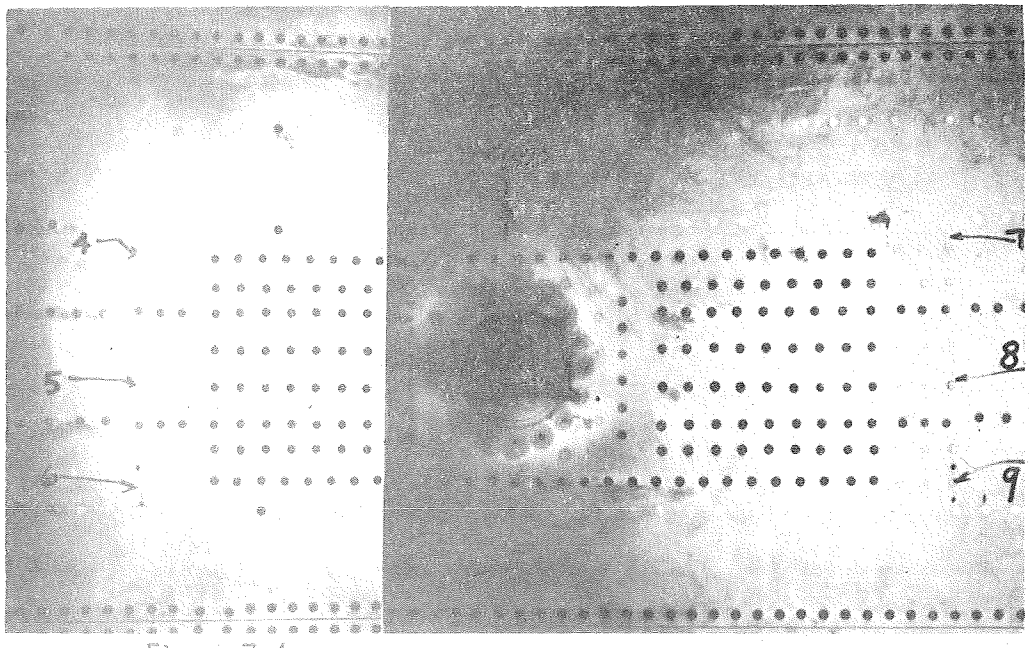


Figure 7-4 Initial Damage
Cracks 4, 5 and 6 were located in the doubler run out area at WS 105.5R.
Cracks 7, 8, 9 and 10 were located in the doubler runout area at WS 135.3R.

RK 0259-6

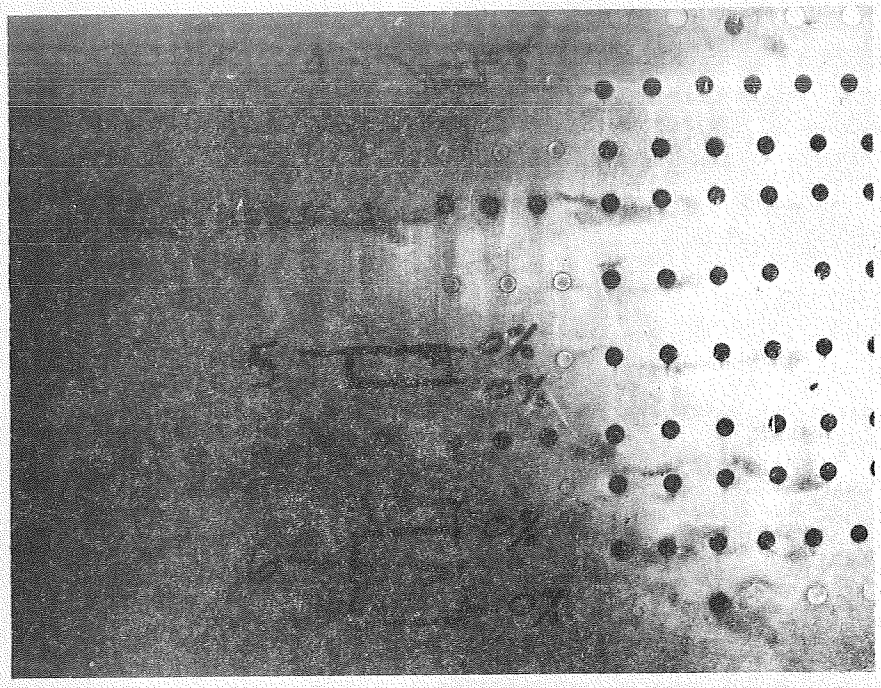


Figure 7-5 Propagation
Crack 6 extended fwd into a fastener hole at 97.7% of limit load.

RK 0253-4

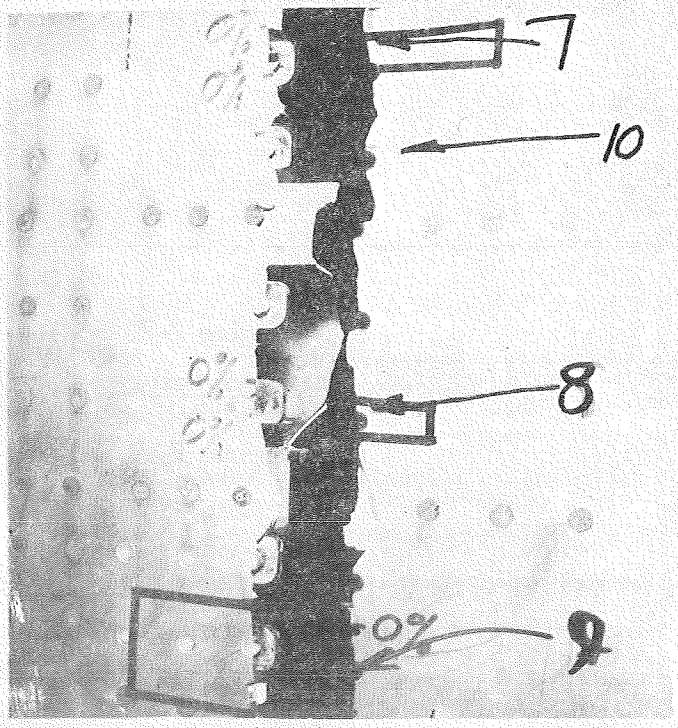
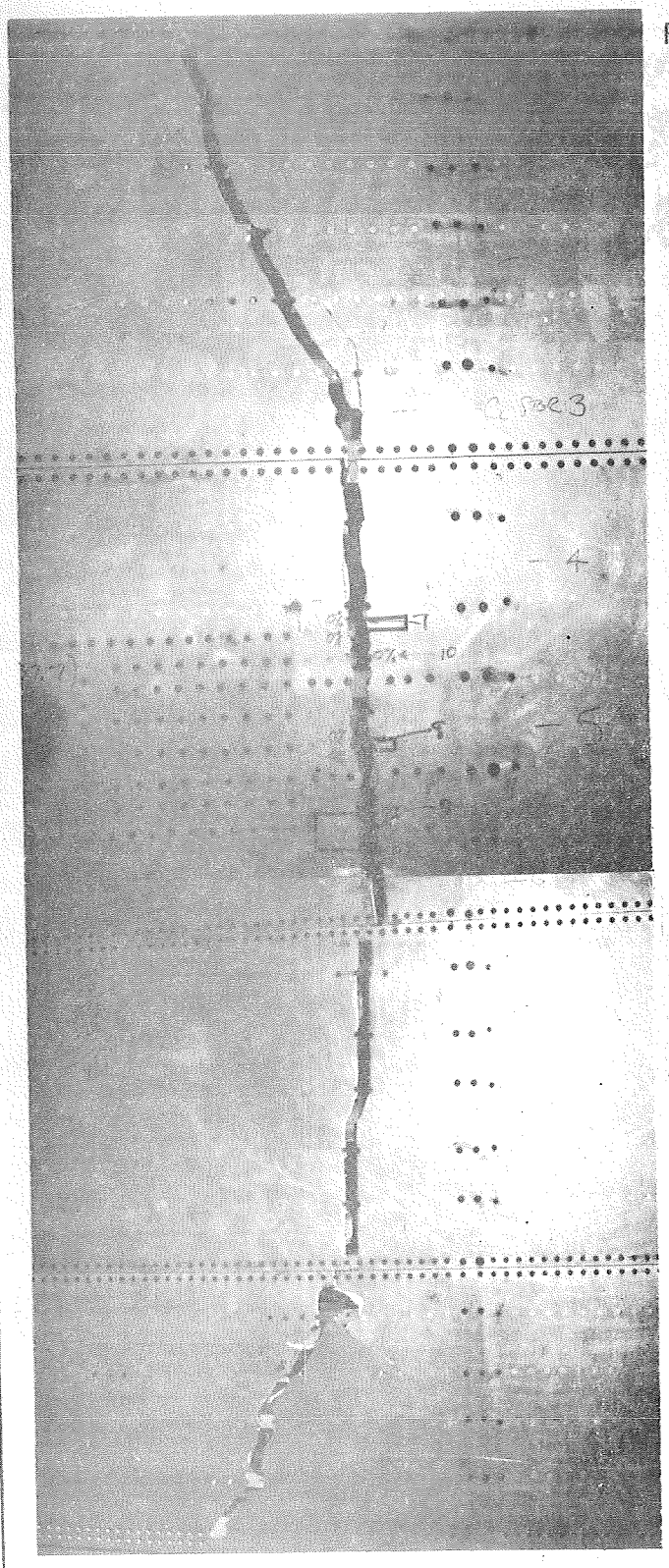


Figure 7-6 Final Failure
The final failure ran through cracks 7, 8, 9 and 10 at WS 135.3R.



RK 0259-4

RK 0259-1

Figure 7-7

Final Failure

Final failure occurred through cracks 7, 8, 9 and 10 at WS 135.3R.

RK 0259-7

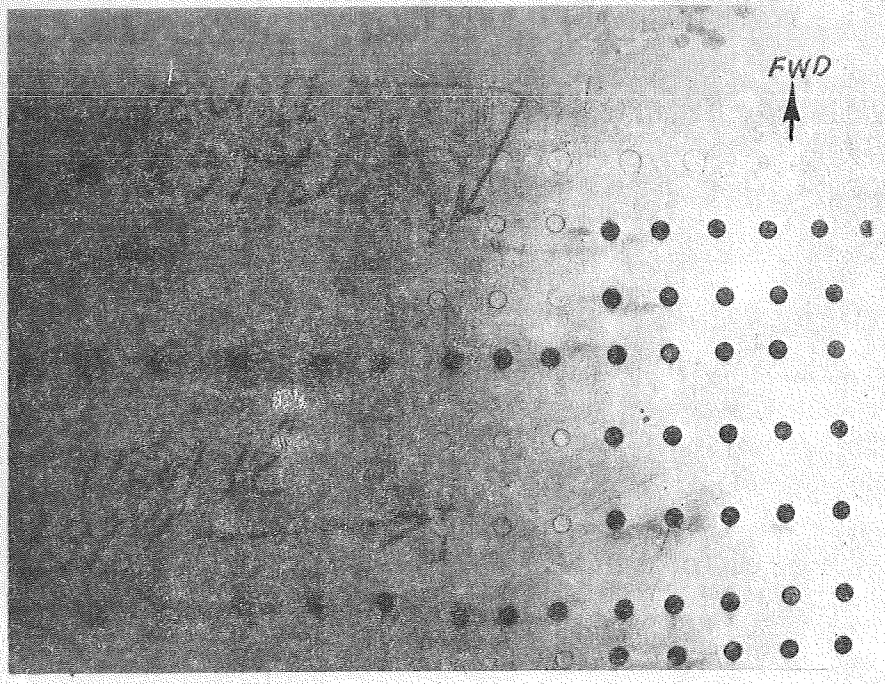


Figure 7-8 New Damage
Cracks 11 and 12, located at WS 135.3L in the doubler run out area were found after test completion at 97.7% limit load.

RK 0259-3

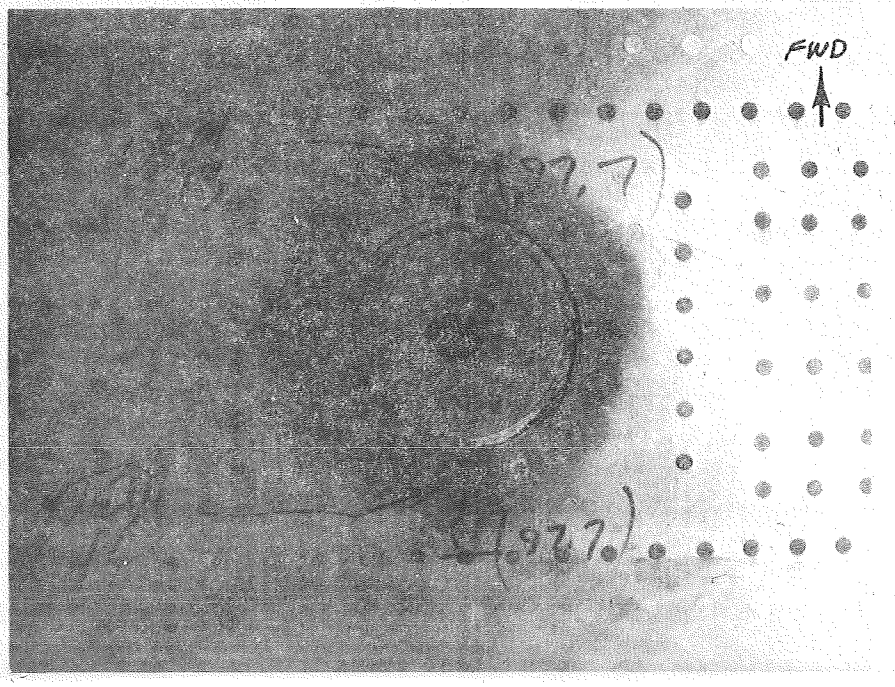


Figure 7-9 New Damage
Cracks 13 and 14 were found after applying 97.7% of limit load in the fuel filler cutout area at WS 120.5R.

PART 8

Specimen 8

RESIDUAL STRENGTH
DOWNBENDING TEST RESULTS

TEST RESULTS

Principal initial cracking was at the corners of the door cutout in the vicinity of W.S. 182 and W.S. 200. Largest of these cracks was 1.4 inches in length, and all of them had one terminal at the edge of the cutout and the other terminal at a fastener hole, as shown in Figures 8-1, 8-2 and 8-3.

Loading was uneventful to 100%, at which point the load level was decreased from 100% to 20% for inspection. No damage was detected. Loading was increased to 105%, when a small sonic report was heard. Loading was immediately decreased from 105% to 20% for inspection, but no damage was detected. Loading was resumed and proceeded uneventfully to failure at W.S. 61(R) at 117% Limit Load, without any interim sonic reports. No damage was found at this location prior to failure because practice on this and all previous test specimens (except #7) was to remove elastomeric coating only in the vicinity of the reported and/or anticipated locations with fatigue cracking. Thus, no initial cracks at the eventual failure location (W.S. 61(R) would have been visible, because of the presence of the elastomeric coating, unless these cracks were at least several inches in length.

Post-failure examination of the fracture surfaces from Test Specimen #8 revealed several small fatigue cracks under the heads of countersunk fasteners. Effective crack lengths comprised of small cracks extending on both sides of countersunk fastener holes were sufficiently large to propagate catastrophically in the Al 7178-T6 material at the stress levels associated with 117% Limit Load.

Table I-8 gives a summary of crack lengths.

TABLE I-8

SUMMARY OF CRACK LENGTHS, SPECIMEN 8

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Location	Fig. No.	Initial Crack Length, In.	Downbending 2 Residual Strength Test (d) After Failure at 116.6% Limit Load
1	The crack originated in the edge of a fastener hole. The hole, common to the door doubler and skin, was one of a row adjacent to the access door attachment fasteners.	199.8L	0.5" fwd of the stringer 5 ℄ a b	8-1	1.2 0.3 0.4 (e)	1.2
2	Same as crack 1	182.3L	0.5" aft of the stringer 2 ℄ a b	8-2	0.7 0.1(e) 0.3	0.7
3	Same as crack 1	182.3R	0.5" aft of the stringer 2 ℄ a b	8-3	1.4 0.6 0.3	1.4
4	Cracks originated in the edges of fastener holes common to the skin and the rib	61R	From front to rear beam along the W.S. 61R rib cap			Complete Chordwise Failure

NOTE: Footnotes found in Table III of the General Section, Part G.

RK 0301-2

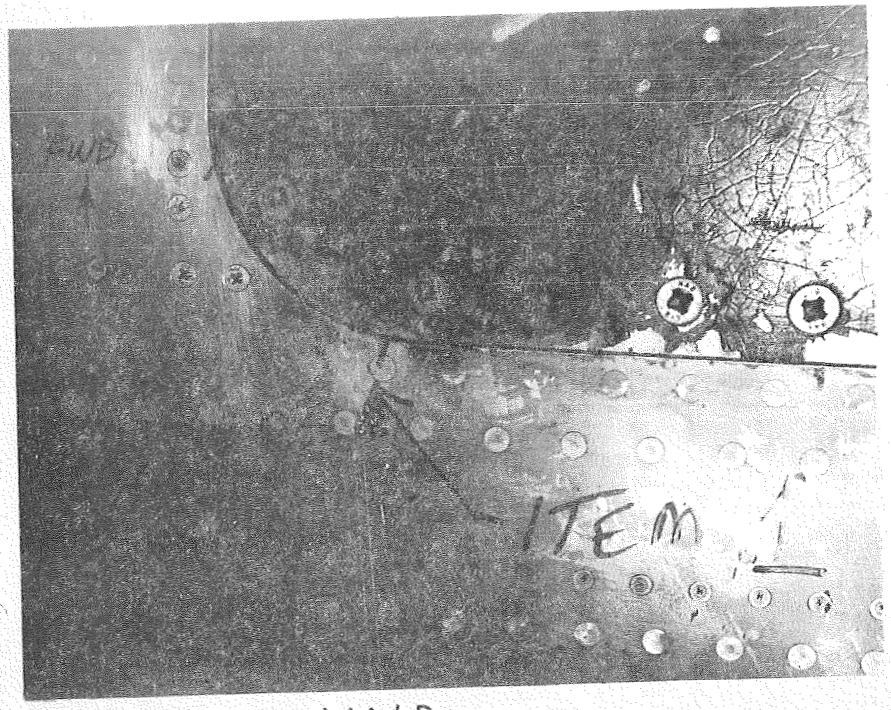


Figure 8-1 Initial Damage
Crack 1 was located at WS 199.8L in the aft outboard corner of the access door cutout.

RK 0301-3

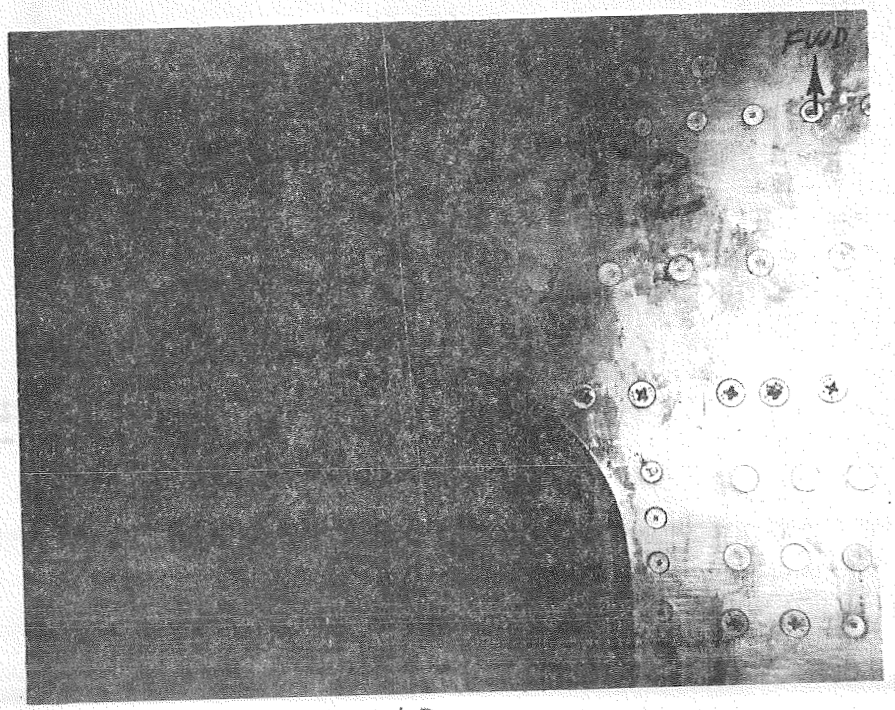


Figure 8-2 Initial Damage
Crack 2 was located at WS 182.3L in the fwd inboard corner of the access door cutout.

RK 0301-1

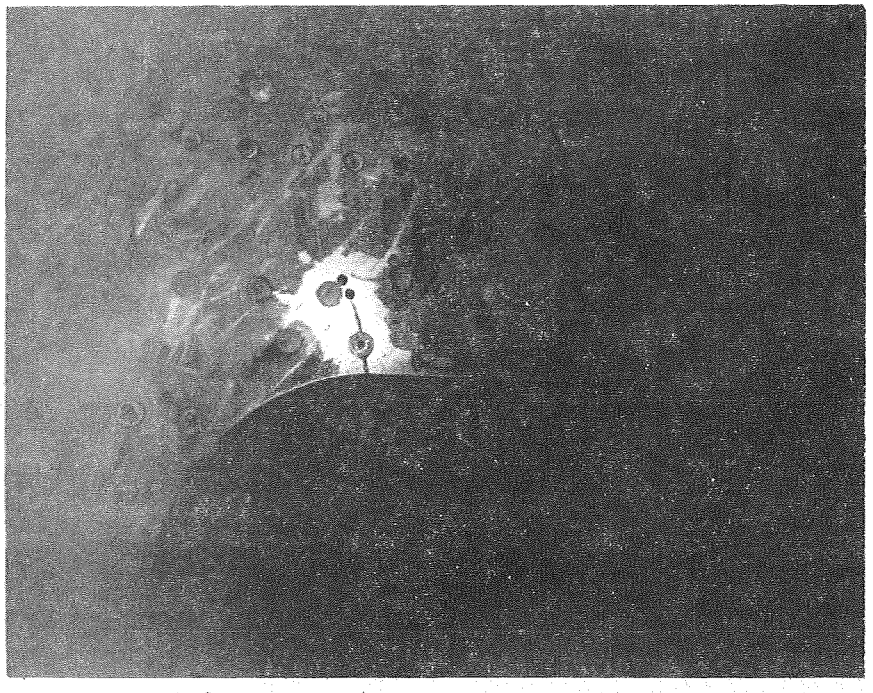


Figure 8-3 Initial Damage
Crack 3 was located at WS 182.3R in the fwd inboard corner of the access door cutout.

PART 9

Specimen 9

CRACK PROPAGATION & RESIDUAL STRENGTH TEST RESULTS

TEST RESULTS

10,000 cycles at 25% Limit Load (downbending) were applied, followed by 10,000 cycles at 22-1/2% Limit Load.

Crack history is shown in Table I-9 for major damage and II-9 for minor damage. Largest initial crack lengths (numbers 1 and 4) were 3.6 inches at W.S. 34.5L (Figure 9-1) and 2.8 inches at W.S. 135R (Figure 9-6). All of the four initial cracks terminated at holes (fastener or stopdrill) except one tip of the 3.6-inch crack which terminated at a skin location that was directly under the flange of a reinforcing hat section (Figure 9-1).

As may be observed from Table I-9, significant crack growth data during the static residual strength test were detected (by sonic reports) at 70%, 74%, 78%, and 81% Limit Load (where failure occurred). At 74% Limit Load, the largest original crack (number 1) of 3.6 inches had grown to only 4.0 inches (W.S. 34.5L), while the 1.4-inch crack (numbers 2 and 11) at the same wing station had grown to 10 inches. After 78% Limit Load, the 3.6-inch crack jumped to 8.2 inches; at 81% Limit Load both of these cracks coalesced and failure occurred at W.S. 34.5L. The original 2.8-inch crack at W.S. 135R only propagated to 2.9 inches at the end of the test.

TABLE I-9

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 9

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Downbending 1, Phase 1 Cyclic Test, R = 0.1 Max. Load = 25% Limit Load — Number of Cycles (d): 0	4000	7500	8000	8500	Phase 2: 13,000
1	The crack originated in the edge of a last fastener hole common to the internal reinforcing doubler of the access door cutout and the skin	34.5L	The stringer 5 ℄	9-1	3.6					3.6
			a	9-3	1.1					1.1
			b		0.8					0.8
2	Same as crack 1		0.5" fwd of the stringer 2 ℄	9-2	0.5					0.5
			a		0.1(e)					0.1
			b		0.1(e)					0.1
11	Same as crack 1		0.5" aft of the stringer 2 ℄		0.9					0.9
			a		0.3(e)					0.3
			b		0.3(e)					0.3
7	Same as crack 1		0.5" fwd of the stringer 3 ℄				0.1			0.1
			a				0.1			0.1
10	Same as crack 1	34.5L	0.5" fwd of the stringer 4 ℄							
			a							
3	Same as crack 1	37.5R	0.5" fwd of the stringer 5 ℄	9-3	0.8				0.8	0.9
			a	9-4	0.5				0.5	0.6
			b		0.1					0.1

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-9 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 9

UPPER SURFACE SKIN, (a)

Crack No.	Downbending 1, Phase 2 Cyclic Test, R = 0.1 Max. Load = 22.5% Limit Load — Cumulative Number of Cycles (d) 14,000	17,000	19,000	19,500	20,000	Downbending 2 Residual Strength Test — After Applying 70% Limit Load	After Applying 74% Limit Load	After Applying 78% Limit Load	After Failure 81% Limit Load
1	3.6				3.6	3.7		8.2	
a	1.1				1.1	1.2		5.6	
b	0.8				0.8	0.8		1.1	
2	0.5			0.5	0.7	3.9	10.4	10.6	
a	0.1			0.1	0.2	1.0	1.0	1.2	
b	0.1			0.1	0.2	2.6	9.1	9.1	
11	0.9				0.9				Complete Chordwise Failure
a	0.3				0.3				
b	0.3				0.3				
7	0.1				0.1	0.1			
a	0.1				0.1	0.1			
10						2.8(b)			
a						2.8			
3	0.9	1.0	1.1	1.1	1.3	2.3		19.8(b)	19.8
a	0.6	0.7	0.8	0.8	0.9	1.1		10.3	
b	0.1			0.1	0.3	0.9		9.2(b)	

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-9 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 9

UPPER SURFACE SKIN (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Downbending 1, Phase 1 Cyclic Test, R = 0.1 Max. Load = 25% Limit Load — Number of Cycles (d) 0	4000	7500	8000	8500	Phase 2 13,000
4	The crack originated in the edge of a last fastener hole common to the internal reinforcing doubler of the fuel filler hole and the skin	135R	2.8" fwd of the stringer 4 ℄ a	9-6	2.8					2.8
			b	9-7	0.8(e)					0.8
					0.8(e)					0.8
5	Same as crack 4	105R	2.8" aft of the stringer 5 ℄ a	9-8		0.2				0.2
			b			0.2				0.2
13	Same as crack 4	105R	2.8" aft of the stringer 5 ℄ a	9-8		0.7				0.7
			b			0.2				0.2
						0.2				0.2

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-9 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 9

UPPER SURFACE SKIN (a)

Crack No.	Downbending 1, Phase 2 Cyclic Test, R = 0.1 Max. Load = 22.5% Limit Load — Cumulative Number of Cycles (d) 14,000	17,000	19,000	19,500	20,000	Downbending 2 Residual Strength Test — After Applying 70% Limit Load	After Applying 74% Limit Load	After Applying 78% Limit Load	After Failure 81% Limit Load
4	2.8		2.8	2.9					2.9
a	0.8			0.8					
b	0.8		0.8	0.9					
5	0.2				0.2			0.2	0.3
a									
b	0.2				0.2			0.2	0.3
13	0.7				0.7				0.7
a	0.2				0.2				
b	0.2				0.2				

NOTE: Footnotes found in Table III of the General Section, Part G.

CA FORM 425A-1

TABLE II-9

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 9

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered (Inches)	(d) Length After 20,000 Cycles	Length After Failure at 81% Limit Load
8	The crack originated in the edge of a last fastener hole common to the internal reinforcing doubler of the fuel filler hole and the skin	105R	0.5" aft of the stringer 5 ℄	9-8	8,500			
			a			0.2	0.2	0.3
			b			0.2	0.2	0.3
6	Same as crack 8	135L	2.8" aft of the stringer 4 ℄	9-9	4,000			
			a			0.5	0.5	0.5
			b			0.1	0.1	0.1
						0.2	0.2	0.2
9	Same as crack 8	135L	0.5" aft of the stringer 5 ℄	9-9	8,000			
			a			0.1	0.5	0.6
			b				0.1	0.1
						0.1	0.1	0.2

NOTE: Footnotes found in Table III of the General Section, Part C.

RK 0344-2

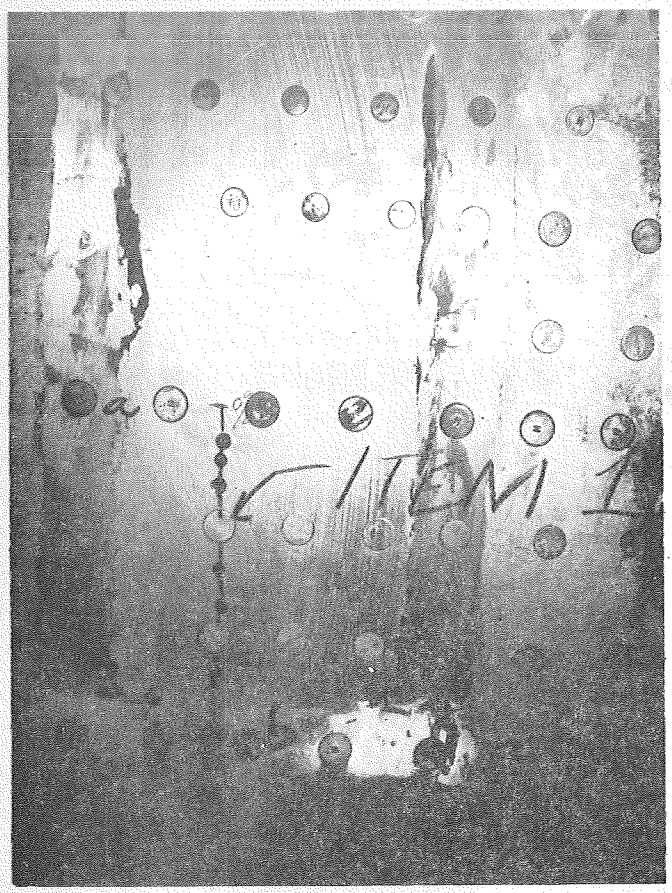


Figure 9-1
Initial Damage
Crack 1 was located at WS 34.5L in the access door doubler run out area.

RK 0344-1

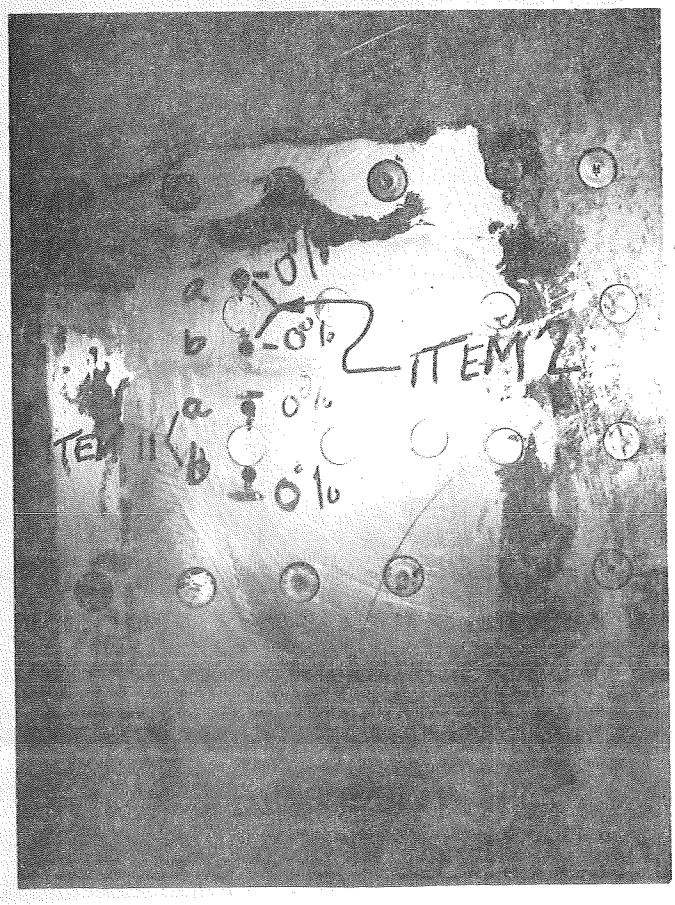


Figure 9-2
Initial Damage
Crack 2 and 11 were located at WS 34.5L. They are in the access door doubler run out area.

RK 0417-1

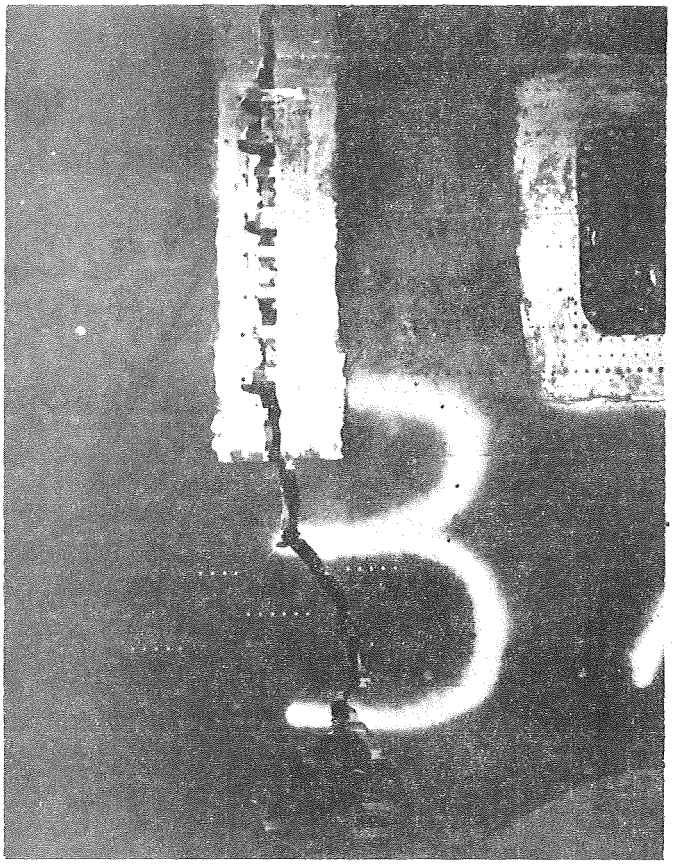


Figure 9-3 Final Failure
The final failure went through cracks 1, 2, 7, 10 and 11 at WS 34.5L.

RK 0344-3

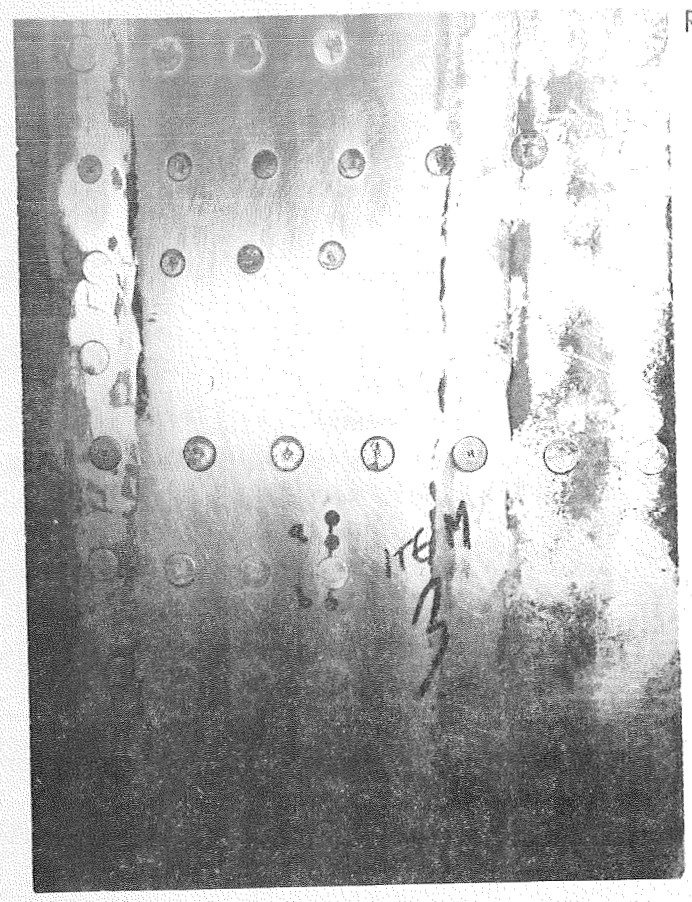


Figure 9-4
Initial Damage
Crack 3 was located in the access door doubler run out area at WS 37.5R.

RK 0391-6

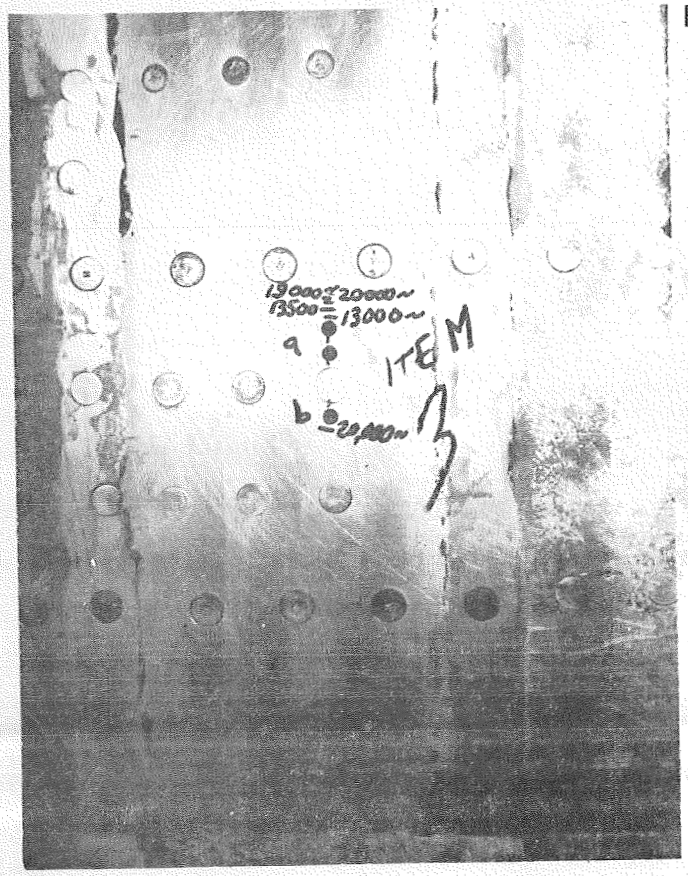


Figure 9-5
Propagation
Crack 3 extended fwd and aft slightly during test at WS 37.5 R.

RK 0344-4

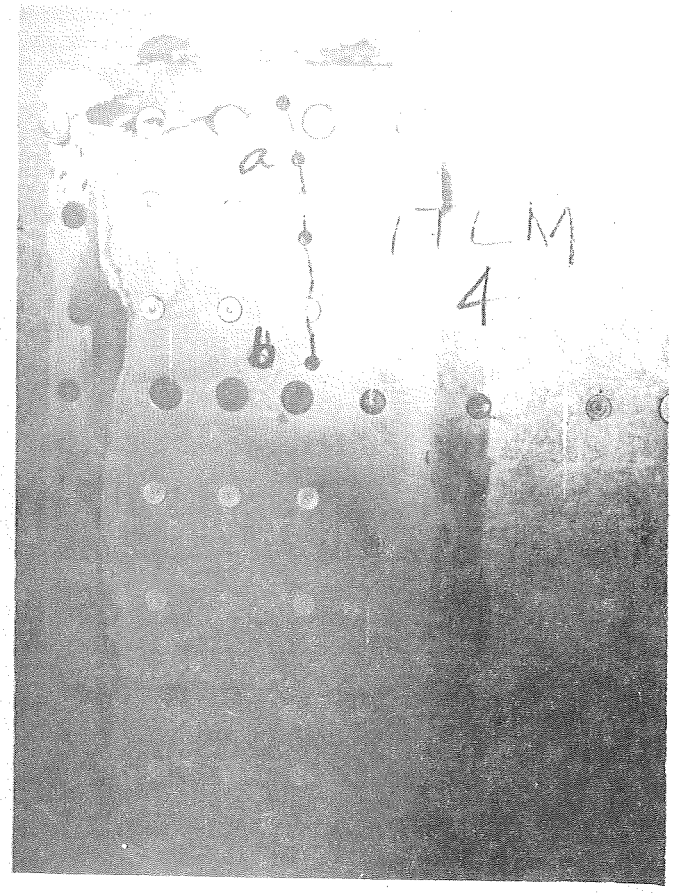


Figure 9-6
Initial Damage
Crack 4 was located in the fuel filler doubler run out area at WS 135R.

RK 0391-4

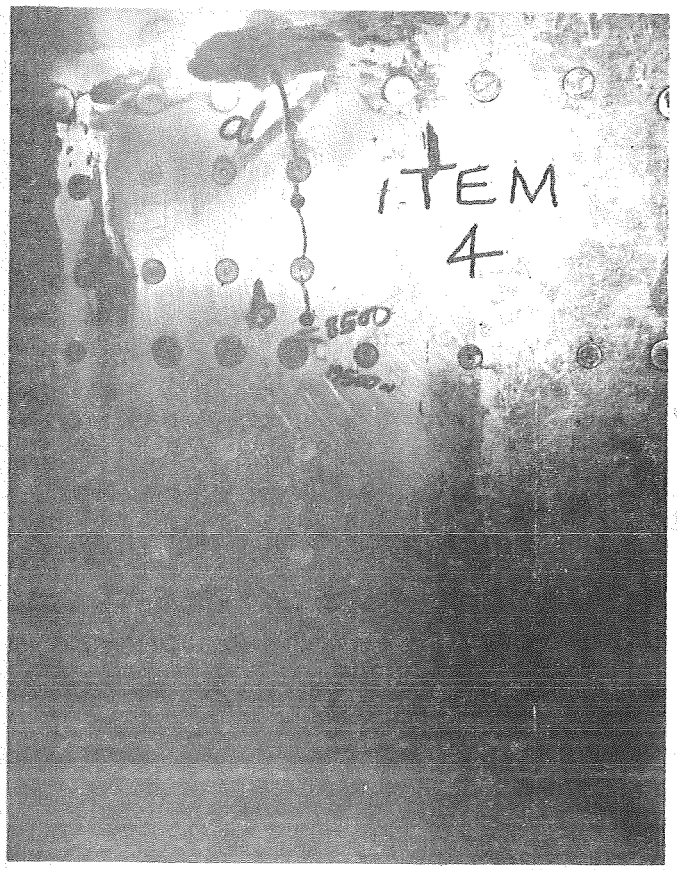


Figure 9-7
Propogation
Crack 4 had slight crack extension on the aft side.

RK 0391-5

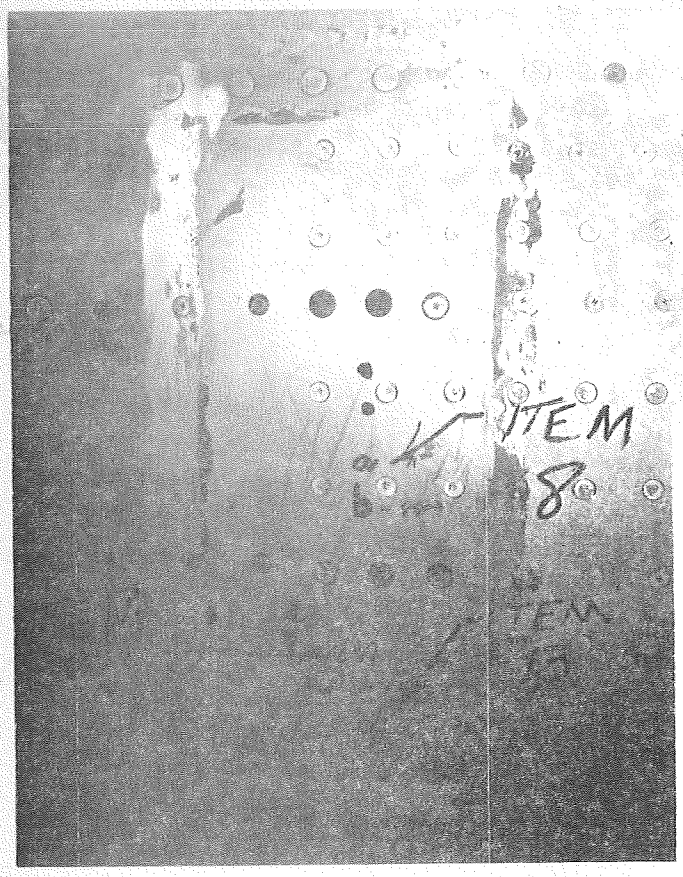


Figure 9-8
Cracks 5, 8 and 13 were located in the fuel filler doubler run out area at WS 105R.

RK 0391-9

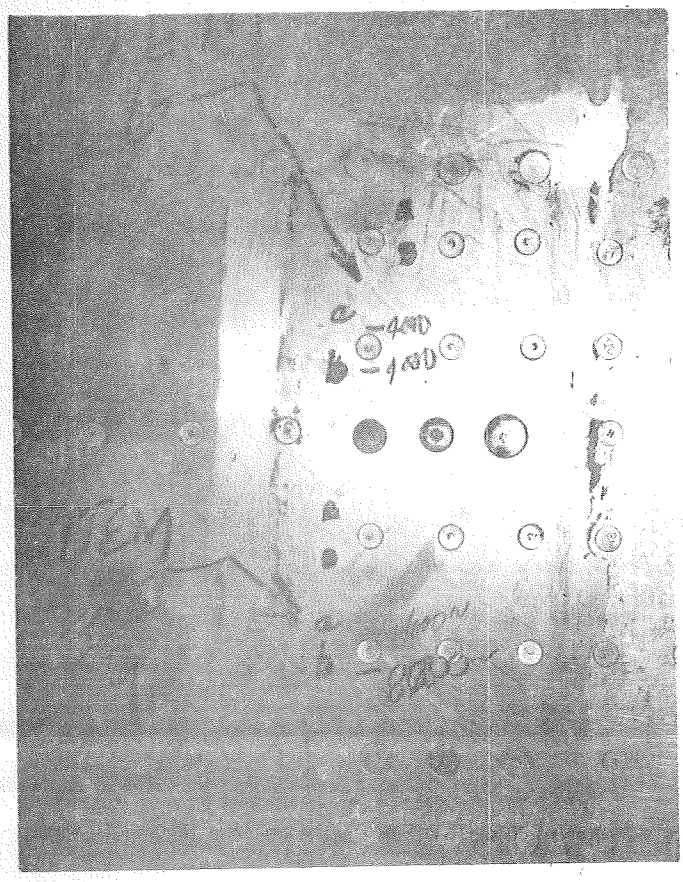


Figure 9-9
Cracks 6 and 9 were located in the run out area of the fuel filler doubler at WS 135L.

PART 10

Specimen 10

CRACK PROPAGATION & RESIDUAL STRENGTH DOWNBENDING TEST RESULTS

TEST RESULTS

10,000 cycles each at 40% Limit Load and 35% Limit Load in downbending were applied. All three of the reported initial cracks were located in the corners of the WS 182.5 - 198.5 door cutout, as shown in Figures 10-7, 10-8, 10-9, 10-10, and 10-11. Largest of these cracks was 1.5 inches (Figure 10-11). The only one of these initial cracks which propagated appreciably was one of the 0.8-inch cracks, crack 2, which grew to 3.0 inches after the static residual strength failure.

All three of these initial cracks had one terminal at the edge of the door cutout and the other terminal at holes (either stopdrilled or fastener). This, along with redistribution of stresses in the vicinity of the crack tips due to reinforcement around the cutout, are regarded as the main reasons why these cracks did not propagate extensively.

Table I-10 gives a summary of major fatigue damage accumulated by the specimen and Table II-10 gives a summary of minor damage.

The static residual strength test to destruction was uneventful to 76% Limit Load, when a loud sonic report was heard. Load was reduced to 20% for inspection. A 13.7-inch crack, number 28, had formed at WS 165R(see Figure 10-14); no prior evidence of cracking at this location had been observed. No substantial growth of other cracks had occurred during static loading to this stage. The next application of loading was stopped at 88% Limit Load after more sonic reports were heard. The 13.7-inch crack, number 28, at WS 165R which occurred at 76% Limit Load had propagated entirely across the panel (19.7 inches), with both crack terminals located at spanwise panel splices. Approximately half of each of two other panels was cracked (number 29 at WS 49.5L, a 9.8-inch crack; and number 7 at WS 174L, a 10.5-inch crack).

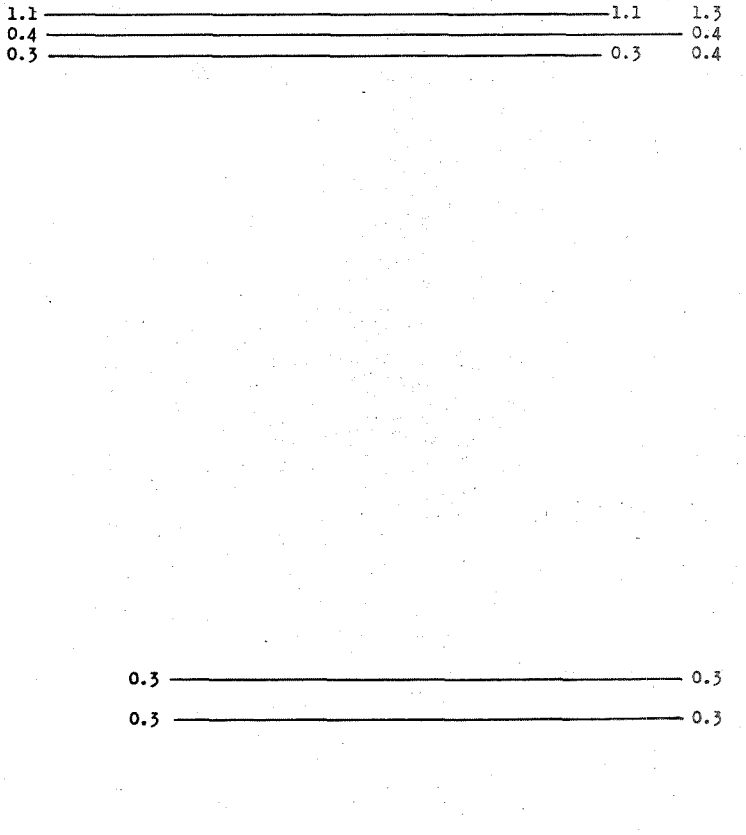
On resumption of loading, total chordwise failure occurred at 87% Limit Load along an irregular fracture line at approximately WS 61L (see Figure 10-9). This was at an indicated 1% lower reading than had been obtained on the preceding load application. Other crack propagation coincident with the failure loading was a 0.5-inch crack, number 11, at WS 61L (not directly in the irregular line of the chordwise failure) which grew to 9.9 inches, and a 9.8-inch crack, number 29, at WS 49.5L grew to 20 inches (entirely across panel).

TABLE I-10
SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 10
UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.
6	Originated in the edge of a fastener hole common to the skin and rib	61L	3.8" fwd of the stringer 1 ℄ a b	10-1 10-2
19	Same as crack 6		2.7" aft of the stringer 1 ℄ a b	
25	Same as crack 6		4" aft of the stringer 1 ℄ b	
24	Same as crack 6		2.7" aft of the stringer 2 ℄	
17	Originated in the edge of a fastener hole common to the skin and stringer flange		1.8" aft of the stringer 3 ℄ a b	
18	Same as crack 17		1.8" fwd of the stringer 4 ℄ a b	10-1 10-2
20	Same as crack 6		4" aft of the stringer 4 ℄	
27	Same as crack 17		2.8" aft of the stringer 6 ℄ a	10-3 10-4
23	Same as crack 6		2.7" fwd of the stringer 10 ℄	
22	Same as crack 6	61L	2.8" fwd of the stringer 11 ℄ b	
11	Same as crack 6	61L	3.8" aft of the stringer 11 ℄ a b	10-3 10-4
29	The crack originated in the edge of the first fastener hole beyond an external repair doubler	49.5L	1.8" aft of the stringer 5 ℄ a b	10-6

Downbending 1, Phase 1
Cyclic Test, R = 0.1
Max. Load = 40% Limit Load

Number of Cycles (d)



NOTE: Footnotes found in Table III of the General Section, Part G.

GA FORM 428A-1

TABLE I-10 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 10

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Downbending 1, Phase 1 Cyclic Test, R = 0.1 Max. Load = 40% Limit Load — Number of Cycles (d)							
					0	3000	3500	4000	4500	6000	7500	8000
2	The skin crack originated in the edge of one of the fastener holes in the row adjacent to the row of access door attachments	182L	0.5" fwd of the stringer 5 ℄	10-7	0.8	1.0		1.0	1.4	1.6	1.7	1.8
			a	10-8	0.3							0.3
			b		0.2(e)	0.4(c)		0.4	0.8	1.0	1.1	1.2
1	Same as crack 2	198.5L	0.5" fwd of the stringer 5 ℄	10-9	0.8				0.8	0.9		0.9
			a	10-10	0.3							0.3
			b		0.2(e)				0.2	0.3		0.3
7	Same as crack 2	174L	0.5" fwd of the stringer 5 ℄	10-8					0.9	1.0	1.1	1.3
			a						0.2	0.2	0.3	0.4
			b						0.3	0.4	0.5	0.6
3	Same as crack 2	199R	0.5" fwd of the stringer 5 ℄	10-11	1.5							1.5
			a		0.3							0.3
			b		0.9(c)							0.9
8	Same as crack 2	182R	0.5" fwd of the stringer 5 ℄									0.7
			a									0.3
			b									0.1
9	Same as crack 6	61R	2.8" fwd of the stringer 1 ℄	10-12								0.2
			a									0.2
			b									
14	The skin crack originated in the edge of a fastener hole common to a stringer flange	131R	1.8" fwd of the stringer 3 ℄	10-13								
28	Same as crack 14	165.4R	1.8" fwd of the stringer 5 ℄	10-14								
			a									
			b									

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-10 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 10

UPPER SURFACE SKIN, (a)

Crack No.	Phase 1		Downbending 1, Phase 2 Cyclic Test, R = 0.1 Max. Load = 35% Limit Load						Downbending 2 Residual Strength Test			
	Cumulative Number of Cycles (d)								After Applying 76% Limit Load	After Applying 84% Limit Load	After Applying 88% Limit Load	After Failure at 87% Limit Load
	9000	10,000	11,500	12,500	14,500	16,000	18,000	20,000				
6	1.4	1.4			1.4	1.5	1.5	1.6	1.6	2.1	2.1	
a	0.4	0.4			0.4	0.5	0.5	0.6	0.6	0.8		
b	0.6	0.6			0.6	0.6	0.6	0.6	0.6	0.8		
19							1.0		1.0	2.4	2.4	
a							0.4		0.4	0.4		
b							0.3		0.3	1.6		
25								0.1			0.1	
b								0.1			0.1	
24								0.2			0.2	
b								0.2			0.2	
17							0.2			0.2	4.0	Complete Chordwise Failure
a											3.4	
b							0.2				0.2	
18							0.2		0.2	1.5	2.2	
a							0.2				0.2	
b										1.7	1.7	
20							0.5				0.5	
a							0.5					
27								0.1			0.1	
a								0.1				
23								0.1			0.1	
b								0.1				
22								0.3			0.3	
b								0.3				
11	0.4						0.4	0.5			0.5	9.9
a												8.6
b	0.4						0.4	0.5			0.5	0.9
29											9.8	20.0
a											6.4	12.7
b											3.2	6.9

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-10 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 10

UPPER SURFACE SKIN, (a)

Crack No.	Phase 1		Downbending 1, Phase 2 Cyclic Test, R = 0.1 Max. Load = 35% Limit Load						Downbending 2 Residual Strength Test			
	Cumulative Number of Cycles (d)								After Applying 76% Limit Load	After Applying 84% Limit Load	After Applying 80% Limit Load	After Failure at 87% Limit Load
	9000	10,000	11,500	12,500	14,500	16,000	18,000	20,000				
2	1.8	1.9	1.9	2.1	2.2	2.3	2.4	2.5	2.5	2.8	3.0	3.0
a	0.3	———	———	———	———	———	———	———	———	———	0.3	
b	1.2	1.3	1.3	1.5	1.6	1.7	1.8	1.9	1.9	2.2	2.4(c)	
1	1.0	1.1		1.1	1.2	1.2	1.3		1.3	1.4		1.4
a	0.3	———	———	———	———	———	———	———	———	0.5		
b	0.4	0.5			0.6	0.6	0.7		0.7	0.8		
7	1.4	1.5	1.7	1.9	2.1	2.3	3.1	3.5	3.8	5.1	10.5	10.5
a	0.5	0.5	0.6	0.7	0.8	0.8	1.2	1.4	1.5	1.5	1.5	
b	0.6	0.6	0.7	0.9	1.0	1.1	1.6	1.7	1.9	3.3	8.7	
3	1.5	———	———	———	———	———	———	———	———	———	1.5	1.5
a	0.3	———	———	———	———	———	———	———	———	———	0.3	
b	0.9	———	———	———	———	———	———	———	———	———	0.9	
8	0.8	0.8	0.9	1.1	1.2	1.3	1.4		1.4	1.7	———	1.7
a	0.3	———	———	———	———	———	———	———	———	0.3		
b	0.2	0.2	0.3	0.5	0.6	0.7	0.8		0.8	1.1		
9	0.2	———	0.2	0.8	———	———	———	———	———	———	0.8	1.0
a	0.2	———	———	———	———	———	———	———	———	———	0.2	0.4
b				0.1	———	———	———	———	———	———	0.1	0.2
14							1.3	1.3	1.6	———	———	1.6
a							0.6	0.6	0.7			
b							0.3	0.3	0.4			
28									13.7	13.7	19.7	19.7
a									3.1	3.1	9.1	
b									10.3	10.3	10.3	

NOTE: Footnotes found in Table III of the General Section, Part G.

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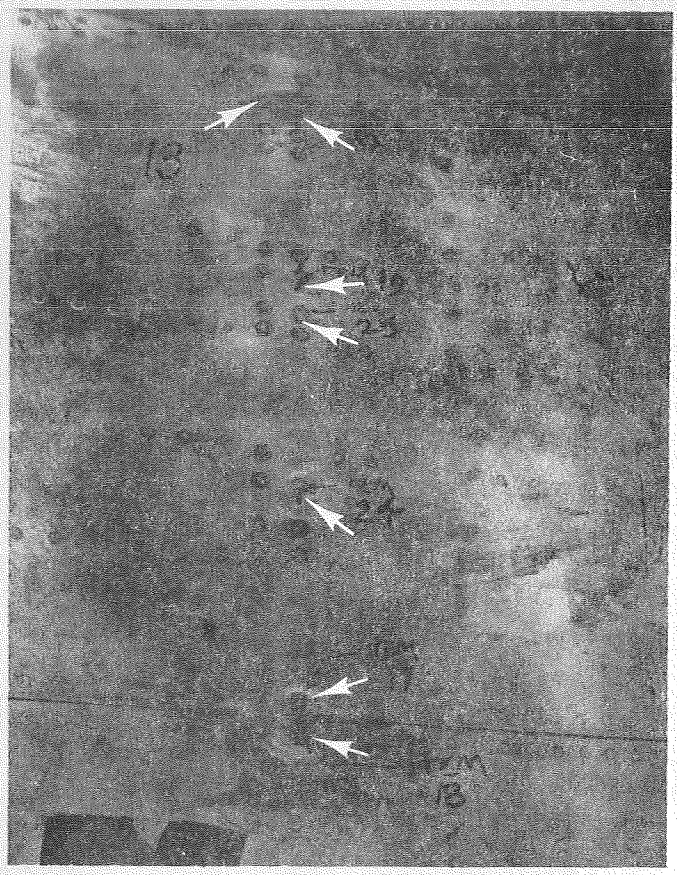
TABLE II-10

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 10

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered (Inches)	(d) Length After 20,000 Cycles	Length After Failure at 87% Limit Load
4	The skin crack originated in the edge of a fastener hole common to a stringer flange	174L	1.8" fwd of the stringer 2 ℄		4,500	0.1	0.6	0.6
			a				0.1	
			b			0.1	0.3	
5	The skin crack originated in the edge of a fastener hole common to the spar cap	179R	4.8" fwd of the stringer 1 ℄	10-15	3,000	0.7		0.7
			a			0.3		
			b			0.1		
10	Same as crack 4	212.5L	3" fwd of the stringer 5 ℄	10-16	8,500	0.3	0.6	0.6
			b			0.3	0.6	
12	The skin cracked in the edge of a fastener hole common to the rib	62R	3.8" fwd of the stringer 1 ℄	10.-2	12,500	0.2	0.7	0.7
			a			0.2	0.2	
			b				0.1	
13	Same as crack 12	62L	3.8" fwd of the stringer 1 ℄	10-1	12,500	0.1		0.1
			b			0.1		
15	The skin cracked in the edge of a fastener hole common to the access door doubler	166.7R	2.8" fwd of the stringer 3 ℄	10-15	18,000	0.1		0.1
			a			0.1		
16	Same as crack 4	131L	2.8" fwd of the stringer 3 ℄	10-17	18,000	0.1	0.7	0.7
			a				0.1	
			b			0.2	0.1	
21	Same as crack 12	61R	3.8" fwd of the stringer 4 ℄	10-12	18,000	0.2		0.2
			a			0.2		
26	Same as crack 4	131R	2" aft of the stringer 5 ℄	10-13	20,000	0.2		0.2
			b			0.2		

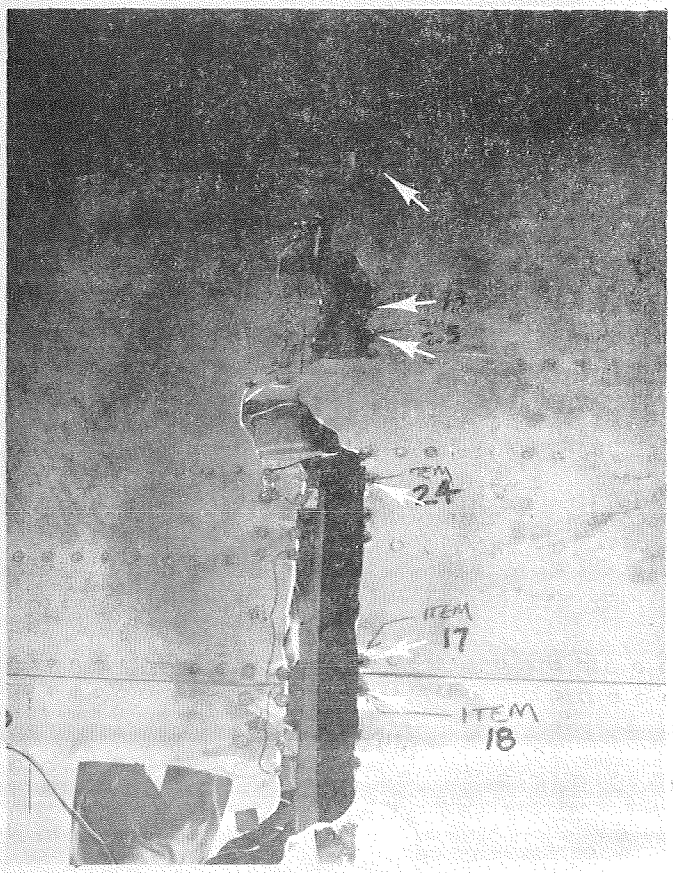
NOTE: Footnotes found in Table III of the General Section, Part G.



RK 0481-8

Figure 10-1

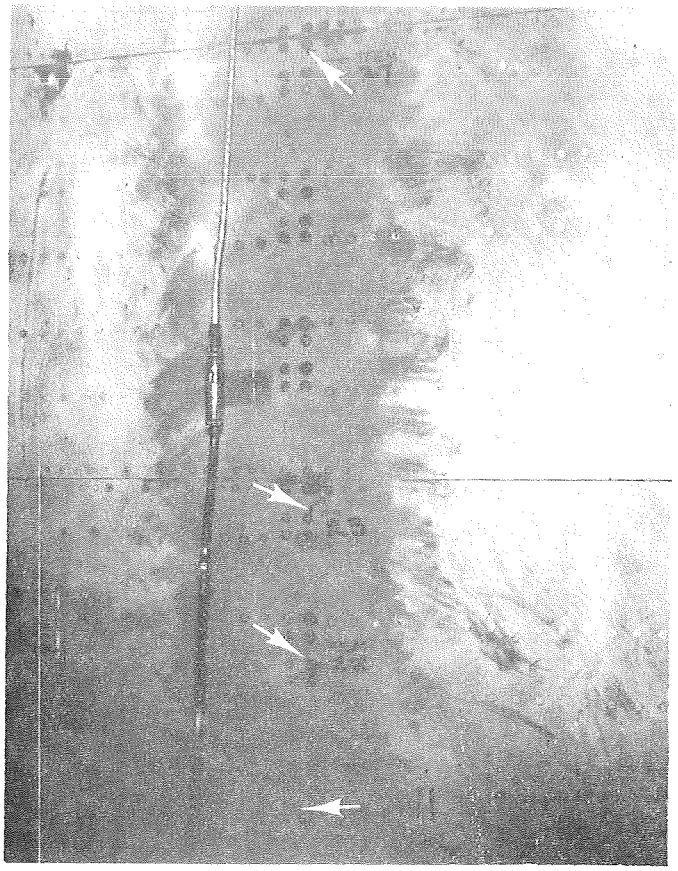
Skin cracks 6, 13, 19, 25, 24, 17 and 18 originated along the line of attachments common to the WS 61L rib. Cracks are as they appeared after completing the propagation test of 20,000 cycles.



RK 0489-6

Figure 10-2

Same area as 10-1 as it appeared after the final chordwise failure at 87% of limit load.

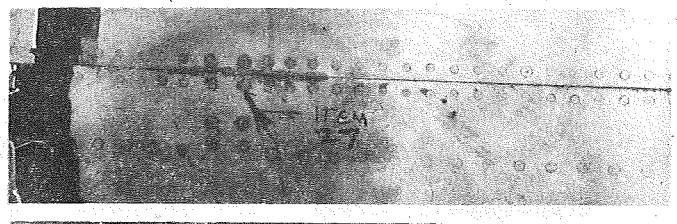


RK 0481-4

Figure 10-3

Skin cracks 11, 22, 23 and 27 originated along the line of attachments common to the WS 61L rib. Cracks are as they appeared after completion of the propagation test of 20,000 cycles.

RK 0489-5



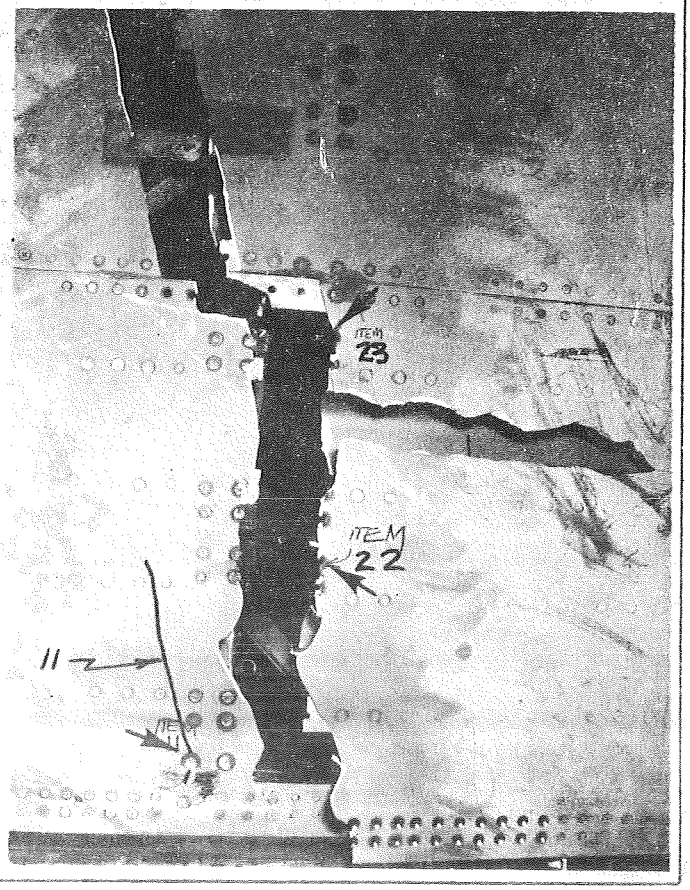


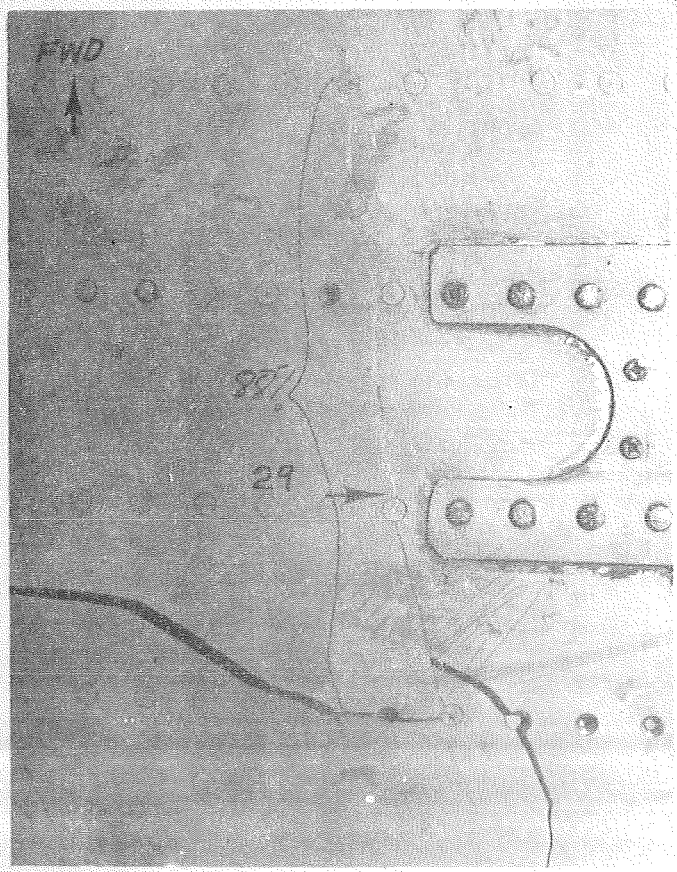
Figure 10-4

Same area as 10-3 as it appeared after the final chordwise failure at 87% of limit load.

RK 0489-2

RK 0489-1

Figure 10-5
Overall view of the chordwise failure at WS 61L which occurred at 87% of limit load.



RK 0489-3

Figure 10-6
Crack 29 occurred at 88% of limit load during the residual strength test.

RK 0417-6

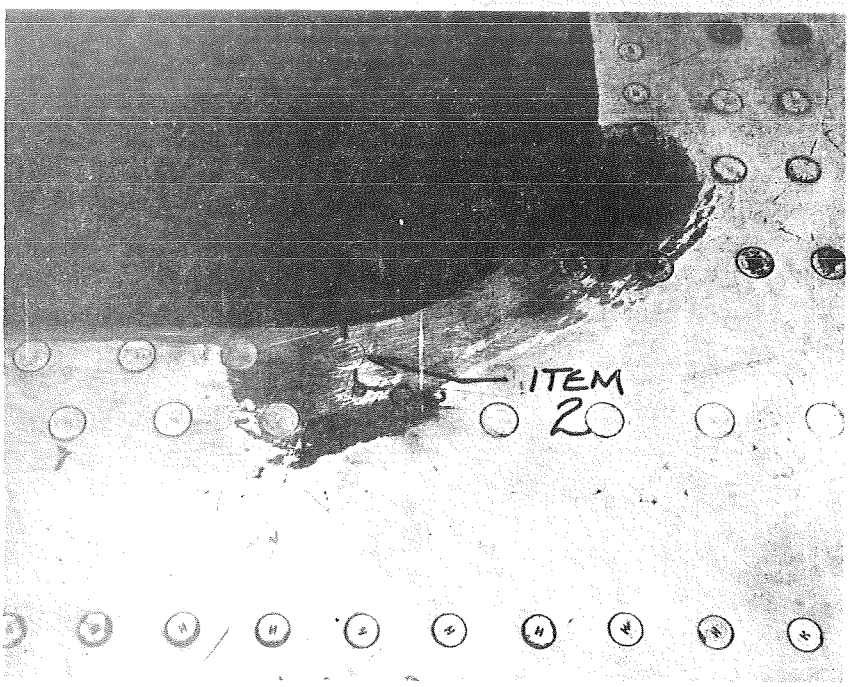
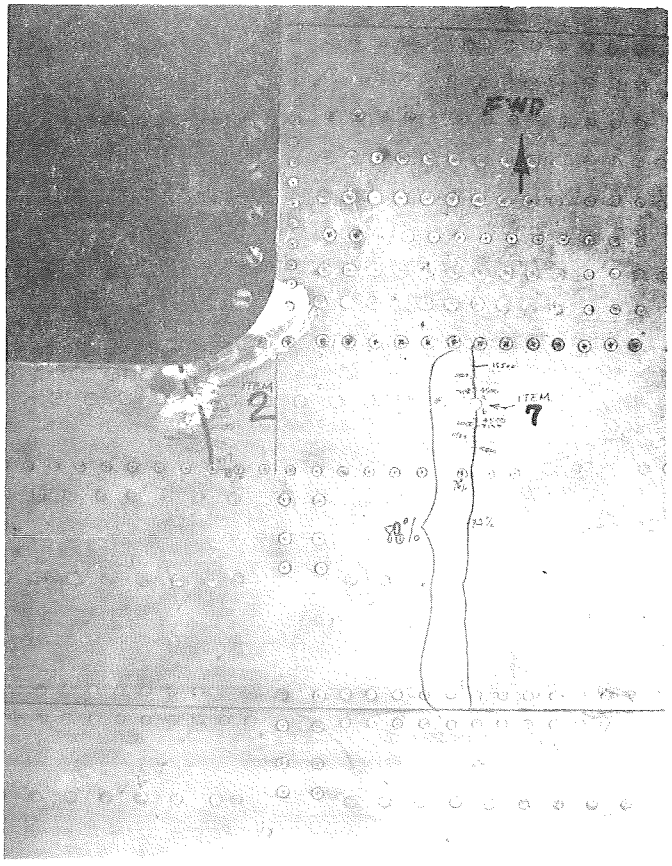


Figure 10-7 Crack 2 in area of the aft inboard corner of the left access door, WS 182L, before testing.



RK 0489-7

Figure 10-8

Same area as above showing cracks 2 and 7 as they appeared after completing the residual strength test.

RK 0417-3

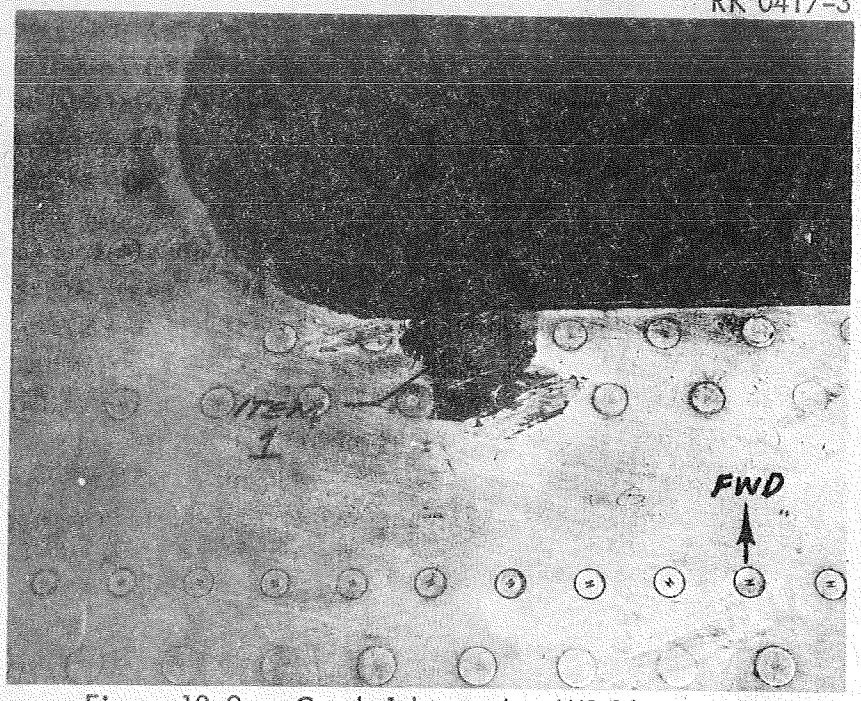


Figure 10-9 Crack 1 located at WS 198.5L in area of the aft outboard corner of the left access door.

RK 0489-8

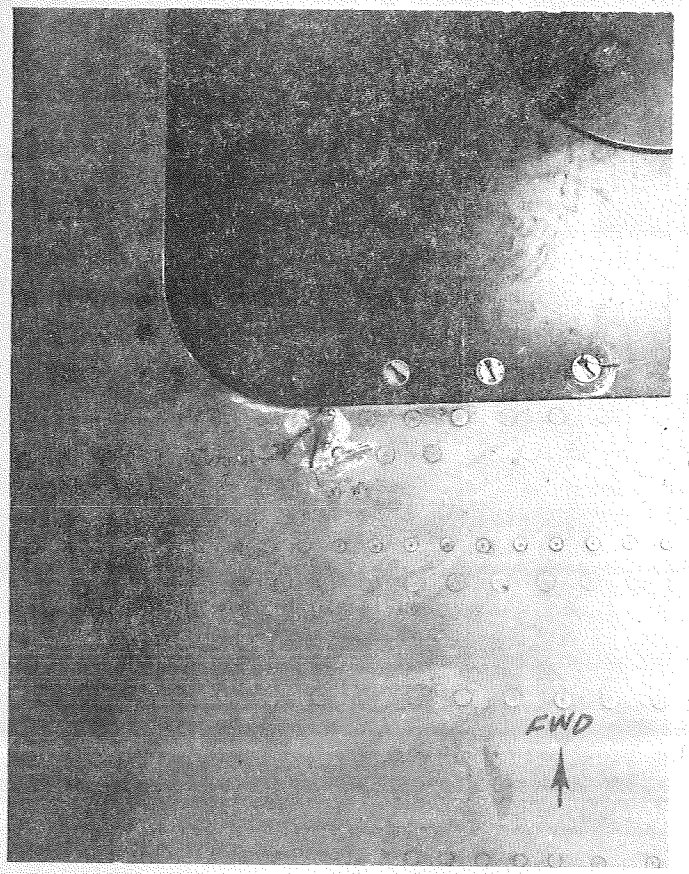


Figure 10-10
Crack 1 after test completion.

RK 0417-5

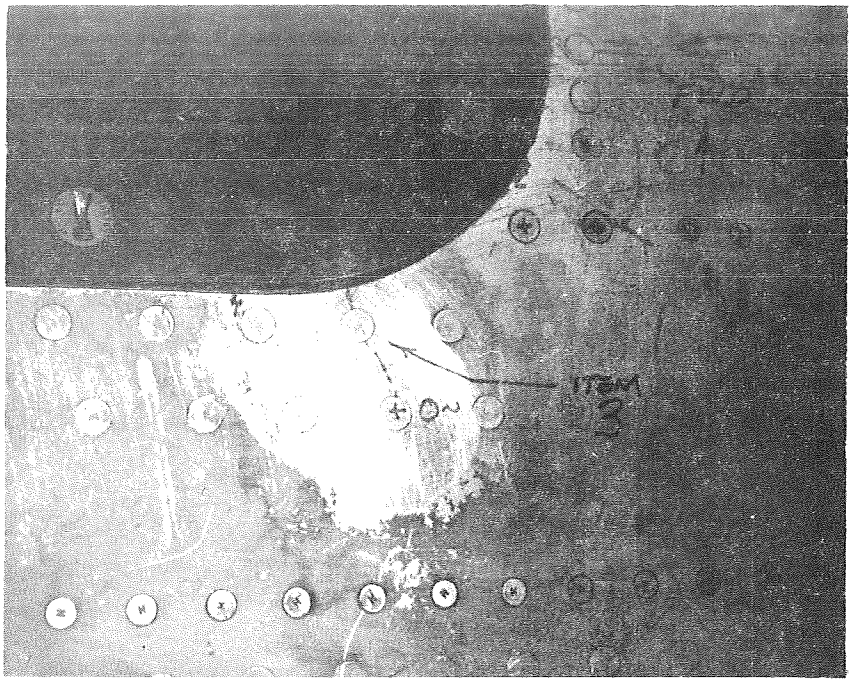


Figure 10-11 Crack 3 was located in the aft outboard corner of the right access door, WS 199R.

RK 0481-9

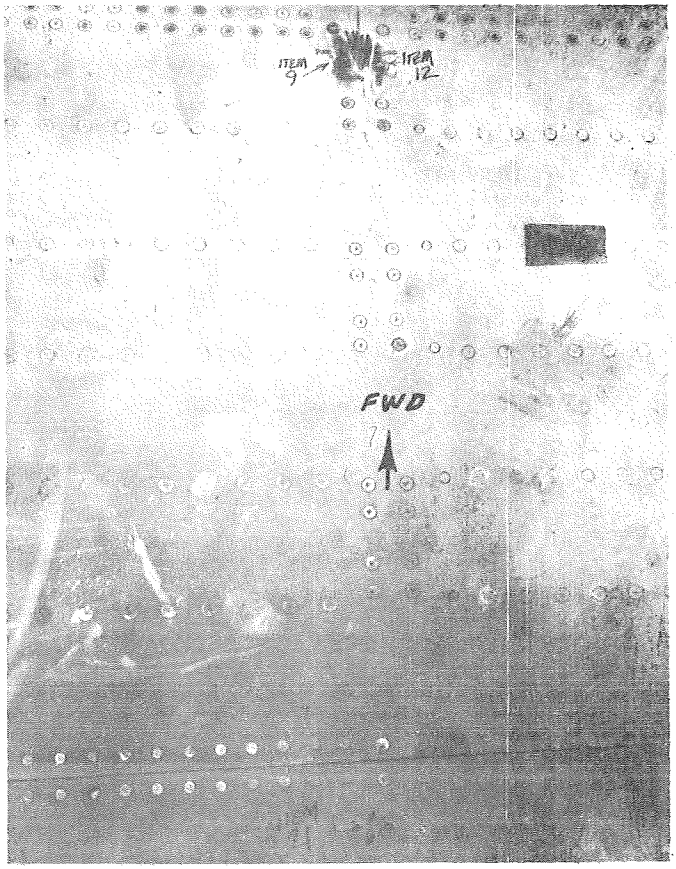


Figure 10-12
Cracks 9, 12 and 21 originated in fastener holes common to the WS 61R rib.

RK 0481-5

ITEM 14

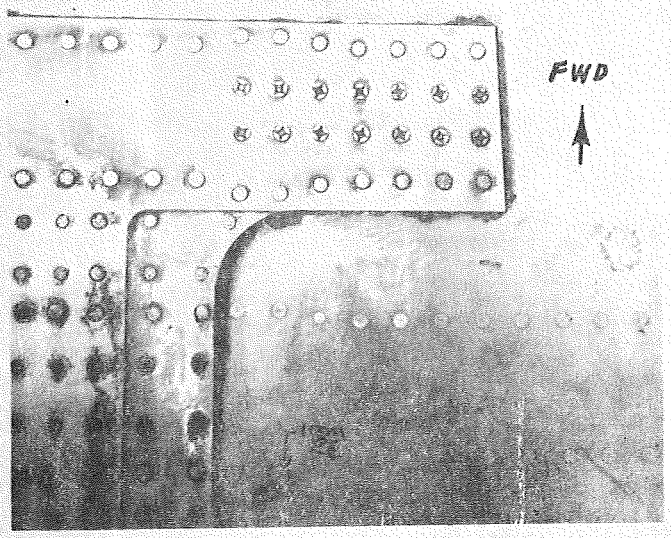


Figure 10-13
Crack 14 originated in a fastener hole common to the fwd flange of stringer 3 at WS 131R.

RK 0489-4

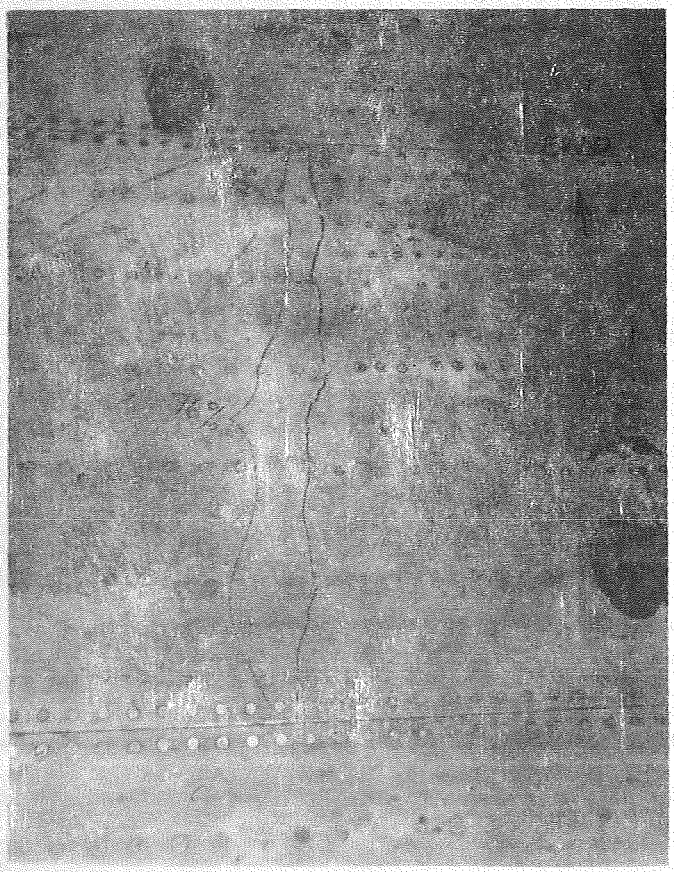


Figure 10-14
Crack 28 originated in a fastener hole common to a stringer flange at WS 165.4R.

RK 0481-6

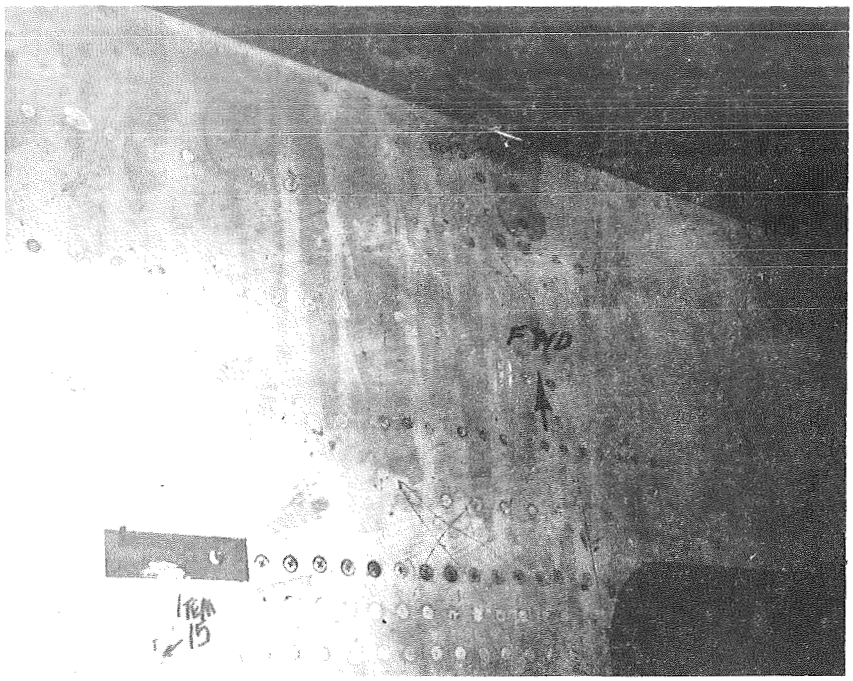


Figure 10-15 Crack 5 originated in a fastener hole common to the spar cap, at WS 179R. Crack 15 originated in a hole common to the access door doubler at WS 166.7 R.

RK 0481-3

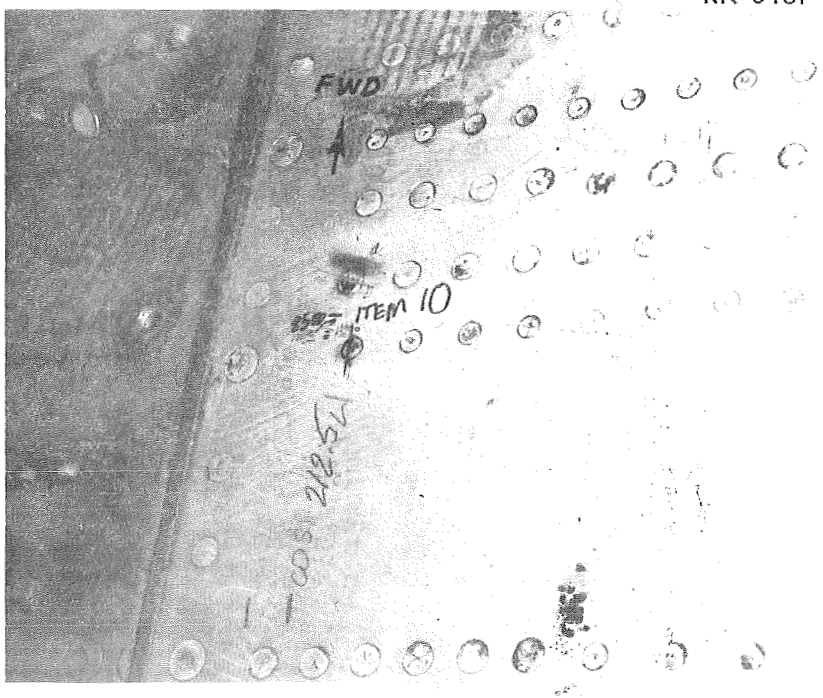


Figure 10-16 Crack 10 originated in a hole common to the fwd flange of stringer 5 at WS 212.5L.

RK 0481-2

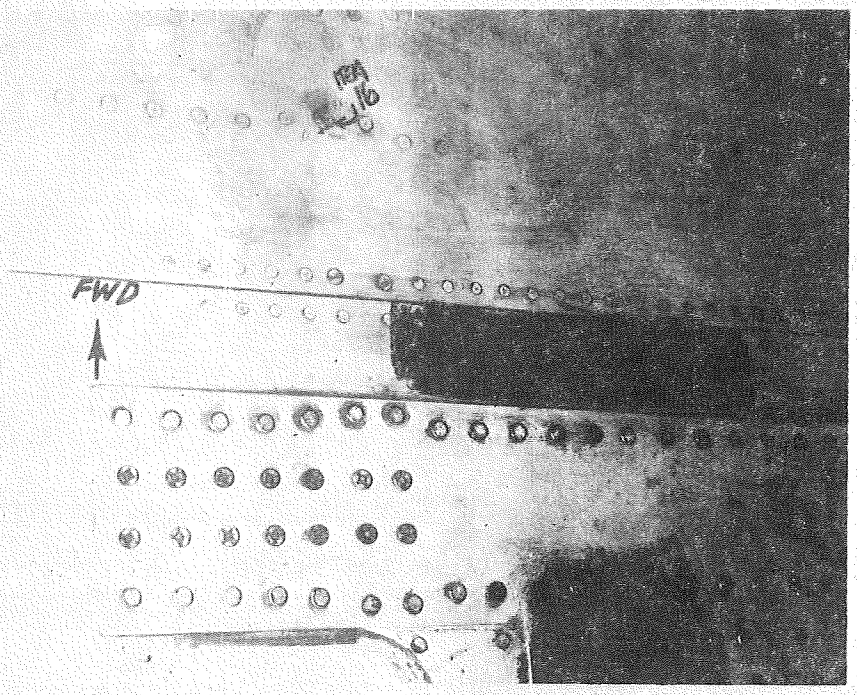


Figure 10-17 Crack 16 originated in a hole common to the fwd flange of stringer 3 at WS 131L

PART 11

Specimen 11

CRACK PROPAGATION & RESIDUAL STRENGTH DOWNBENDING TEST RESULTS

TEST RESULTS

When received, the specimen had 7,660 flight hours logged, and had extensive fatigue preventive repairs. These repairs are shown in Appendix A. The specimen had sustained the following initial damage:

1. Access door cutout corner, WS 182.5L: A 3.0 inch long skin crack, Number 1, at the forward inboard corner of the cutout.

2. Immediately aft of the engine mount, at fasteners common to the spar cap and rib which normally react engine loads, two skin cracks were found, both 0.8 inches long. One of these cracks was found at WS 179.9L, 0.5 inches aft of the forward edge of panel number 1 (crack number 2) and the other at WS 178.5R, 0.5 inches aft of the forward edge of panel number 1 (crack number 3).

Propagation Test

Phase I

This portion of testing consisted of applying cyclic loads which varied from 50% to 5% limit downbending 1 for 10,000 cycles. Damage at the completion of Phase I was not extensive. During Phase I, the initial crack number 1 had grown from 3.0 inches to 4.0 inches, crack 2 had grown from 0.8 inches to 2.1 inches, and crack 3 had grown from 0.8 inches to 1.1 inches. Other significant damage which initiated during Phase I was loacted at WS 213.5R, where a total crack length at the end of Phase I was 4.0 inches; and at WS 214.5L, where a total crack length of 1.4 inches had developed at the end of Phase I.

Phase II

This portion of testing consisted of applying cyclic loads which varied from 40% to 4% limit downbending 1. 4,500 of these cycles (14,500 total cycles) had been applied when cycling was discontinued in order to prevent total specimen failure during the crack propagation test. The principal damage at this point was in the WS 214R area where the total skin crack length had propagated to a length of 33.5 inches and the stringer 'hat' sections for Stringers 7, 8, 9, 10, and 11 had broken. Other serious damage that occured in this area after the crack propagation test is shown in Figures 11-1 and 11-3. The WS 214L area had a total skin crack length of 15.8 inches. Damage in this area after crack propagation testing is shown in Figures 11-9 and 11-10. Other damage was comparatively limited in extent.

Static Residual Strength Test

Static Residual Strength Testing was then conducted under customary control procedures, and progressed uneventfully until failure at 57.3% of limit downbending 2 condition (which is roughly equivalent to 43.4% downbending 1 condition at WS 214). Complete chordwise failure occurred in the WS 214R area. The only other significant propagation to occur was at WS 178.5R where crack no. 3, which was 1.7 inches long prior to the residual strength test propagated to a length of 10.3 inches during the static test.

In evaluating the propagation and failure of Specimen No. 11, it should be noted that many of the fatigue sensitive areas (WS 105L and R, WS 135L and $, WS 34.5L and WS 37.5R) were extensively repaired so as to prevent any static or fatigue crack propagation.

Table I-11 gives a summary of major fatigue damage accumulated by the specimen and Table II-11 gives a summary of minor fatigue damage.

TABLE I-11

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Downbending 1, Phase 1 · Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d) 0	2000	4000	5000	6000	7000	8000	9000	10,000
16	The crack originated in the edge of a fastener hole common to the skin and the W.S. 220 rainbow fitting	213.5R	1.8" aft of the stringer 5 ℄	11-3									0.1
			a	11-4									0.1
			b										-
11	Same as crack 16		2.7" fwd of the stringer 6 ℄	11-3								0.2	
			a	11-4								0.2	
			b	11-6								-	
15	Same as crack 16		1.8" fwd of the stringer 6 ℄	11-3									0.1
			a	11-4									0.1
			b	11-15									-
14	Same as crack 16		2.8" fwd of the stringer 7 ℄	11-3									0.2
			a	11-4									0.2
			b	11-6									-
54	Same as crack 16		2.8" aft of the stringer 6 ℄	11-3									
				11-4									
				11-6									
57	Same as crack 16		2.8" aft of the stringer 7 ℄	11-3									
			a	11-4									
			b	11-6									
9	Same as crack 16		2.7" fwd of the stringer 10 ℄	11-1							0.2	0.4	3.6
			a	11-2									1.5(b)
			b	11-6							0.2	0.4(c)	1.8
12	Same as crack 16		1.8" aft of the stringer 10 ℄	11-1									
			a	11-2									
			b	11-6									
10	Same as crack 16	213.5R	2.7" aft of the stringer 10 ℄	11-1									
			a	11-2									
			b	11-6									
13	Same as crack 16	214.5R	0.5" aft of the stringer 9 ℄	11-1									
			a	11-2									
			b	11-6									

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-11 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN, (a)

Crack No.	Downbending 1, Phase 2 Cyclic Test, R = 0.1, Max. Load = 40% Limit Load — Cumulative Number of Cycles (d)						Downbending 2 Residual Strength Test
	11,000	12,000	13,000	14,000	14,250	14,500	After Failure at 57.3% Limit Load
16	0.1		0.1	0.2			
a	0.1		0.1	0.2			
b							
11	0.8	1.1	1.2	1.3	4.5	5.2	
a	0.3	0.3	0.4	0.5	2.4	3.2	
b	0.2	0.5		0.5	1.7	1.7	
15	0.1						
a	0.1						
b							
14	0.2	0.3	0.8	1.0	1.6	2.5(m)	
a	0.2	0.3	0.3	0.4	0.7	1.5(b)	
b			0.2	0.3	0.6(c)	0.6	
54						0.5	Complete Chordwise Failure
a						-	
b						0.5	
57						0.1(m)	
a						0.1	
b							
9	4.4	5.9	17.3			17.3(m)	
a	1.5	1.5	1.5			1.5	
b	2.5(c)	4.0	15.4(b)			15.4	
12	1.7	4.2					
a	0.7	1.0					
b	0.6	2.7					
10	9.0	9.0					
a	3.2	3.2					
b	5.4	5.4					
13	2.2	2.5	2.6	2.7	7.9	7.9(m)	
a	1.7	2.0	2.1	2.2	7.5	7.5	
b	0.2					0.2	

NOTE: Footnotes found in Table III of the General Section, Part G.

GA FORM 6282A-1

TABLE I-11 (Cont.)
SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 11
UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Downbending 1, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d)								
					0	2000	4000	5000	6000	7000	8000	9000	10,000
5	The crack originated in the edge of a fastener hole. The hole, common to the door doubler and skin, was one of a row adjacent to the access door attachment fasteners	182.5R	0.5" aft of the stringer 2 ℄	11-7		0.1						0.1	0.6
			a	11-8									0.1
			b			0.1						0.1	0.2
34	Same as crack 5	182.5R	0.6" fwd of the stringer 5 ℄	11-8									
			a										
			b										
3	The crack originated in the edge of a fastener hole common to the spar cap, rib and skin panel #1	178.5R	0.5" aft of fwd edge of panel #1	11-7	0.8(c)		0.8	1.1					1.1
			a		0.3								0.3
			b		0.2		0.2	0.5					0.5
17	The crack originated in the edge of a fastener hole common to the stringer leg and skin	140.6R	1.6" fwd of the stringer 7 ℄	11-11									
			a										
			b										
38	The crack originated in the edge of a fastener hole adjacent to an external skin repair doubler	48.7L	1.8" fwd of the stringer 5 ℄	11-12									
			a										
			b										
22	The crack originated in the edge of a fastener hole common to the rib, the stringer leg and the skin.	60.5L	1.8" fwd of the stringer 3 ℄	11-14									
			a										
			b										
80	Same as crack 17	166.8L	1.8" fwd of the stringer 5 ℄	11-13									
			a										
			b										
2	Same as crack 3	179.9L	0.5" aft of the fwd edge of panel #1		0.8(c)	0.8	1.1				1.1	2.1(c)	2.1
			a		0.3								0.3
			b		0.2	0.2	0.5				0.5	1.5	1.5

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-11 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN, (a)

Crack No.	Downbending 1, Phase 2 Cyclic Test, R = 0.1 Max. Load = 40% Limit Load — Cumulative Number of Cycles (d) 11,000	12,000	13,000	14,000	14,250	14,500	Downbending 2 Residual Strength Test After Failure at 57.3% Limit Load
5	0.7	0.7	0.8	0.9		0.9	1.4
a	0.1						0.1
b	0.3	0.3	0.4	0.5		0.5	1.0
34		0.6	0.9	1.1	1.2	1.3	1.3
a		0.1	0.3(b)			0.3	
b		0.1	0.2	0.5	0.6	0.7	
3	1.1	1.5	1.5	1.6	1.6	1.7	10.3
a	0.3						0.3
b	0.5	0.9	0.9	1.0	1.0	1.1	10.0
17	0.2	0.3	0.9	1.0		1.0	1.1
a	0.2	0.3	0.3	0.4			0.4
b	-	-	0.2				0.2
38			0.8	1.0	1.0	1.2	1.2
a			0.2	0.3	0.3	0.4	
b			0.3	0.4	0.4	0.5	
22	0.6	0.7	0.8	0.8	0.9	0.9	1.0
a	0.1	0.1	0.2	0.2	0.3	0.3	0.4
b	0.2	0.3(c)					0.3
80						1.4	1.4
a						0.4	
b						0.6	
2	2.1(c)						2.1
a	0.3						0.3
b	1.5						1.5

NOTE: Footnotes found in Table III of the General Section, Part G.

GA FORM 628A-1

TABLE I-11 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN, (a)

Crack No	Description	Wing Sta. In.	Location	Fig. No.	Downbending 1, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d)								
					0	2000	4000	5000	6000	7000	8000	9000	10,000
1	Same as crack 5	182.5L	0.5" aft of the stringer 2 ℄	11-15									
			a		3.0	3.3	3.4	3.5	3.5	3.6	3.7	3.9	4.0
			b		2.4	2.7	2.8	2.9	2.9	3.0	3.1	3.3	3.4
					0.3	—	—	—	—	—	—	—	0.3
26	Same as crack 5	182.5L	0.5" fwd of the stringer 5 ℄										
			a										
			b										
7	Same as crack 16	214.5L	1.8" fwd of the stringer 5 ℄	11-9									
			a						0.3	0.4	1.2	1.2	1.3
			b						0.3	0.4	0.5	0.5	0.6
											0.3	—	0.3
31	Same as crack 16		2.8" aft of the stringer 6 ℄	11-9									
			a										
			b										
32	Same as crack 16		2.8" fwd of the stringer 7 ℄	11-9									
33	Same as crack 16		1.8" fwd of the stringer 7 ℄	11-9									
8	Same as crack 16	214.5L	1.8" fwd of the stringer 9 ℄	11-10									
			a							0.1	—	—	0.1
			b							0.1	—	—	0.1
41	Same as crack 16	213.7L	1.8" aft of the stringer 7 ℄	11-9									
			a										
			b										
53	Same as crack 16		2.8" aft of the stringer 10 ℄	11-9									
			a										
			b										
40	Same as crack 16	213.7L	1.8" aft of the stringer 10 ℄	11-10									
			a										
			b										

NOTE: Footnotes found in Table III of the General Section, Part G.

GA FORM 423A-1

TABLE I-11 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN (a)

Crack No.	Downbending 1, Phase 2 Cyclic Test, R = 0.1 Max. Load = 40% Limit Load — Cumulative Number of Cycles (d) 11,000	12,000	13,000	14,000	14,250	14,500	Downbending 2 Residual Strength Test — After Failure at 57.3% Limit Load
1	4.0	4.0	4.1				4.1
	3.4	3.4	3.5				3.5
	0.3						0.3
26	1.1		1.1	1.5	1.5	1.7	1.7
a	0.3						0.3
b	0.5		0.5	0.9	0.9	1.1	1.1
7	1.4		1.4	1.5			1.5
a	0.6						0.6
b	0.4						0.4
31	0.7	1.1	3.7	4.6			4.6
a	0.3(b)						0.3
b	0.1	0.5	3.1	4.0			4.0
32	0.8	1.3					
a	0.2	0.7					
b	0.2	0.2					
33	0.9	1.2					
a	0.3	0.3					
b	0.1	0.4					
8	3.6	4.1	5.0	5.6	5.6	5.8	5.8
a	0.9	1.1	1.2	1.4	1.4	1.6	
b	2.2	2.5	3.3	3.7	3.7	3.8(c)	
41			0.3	1.1	1.2	2.5	2.5
a			0.3	0.5	0.6	0.9	
b				0.2	0.2	1.2	
53				0.6	0.8		
a				0.2	0.2		
b				0.1	0.3		
40			0.3	1.1	1.3	1.4	1.4
a			0.3	0.4	0.5	0.6	
b				0.2	0.3	0.4	

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE II-11

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered (Inches)	(d) Length After 14,500 Cycles	Length After Failure at 57.3% Limit Load
35	The crack originated in the edge of a fastener hole common to the skin and the W.S. 220 rainbow fitting	214.5L	1.8" fwd of the stringer 10 ℄	11-10	11,750	0.2	0.4	0.4
			a			0.2	0.4	0.4
49	Same as crack 35	213.7L	2.0" aft of the stringer 5 ℄	11-9	13,000	0.2		0.2
			b			0.2		0,2
47	Same as crack 35		2.7" fwd of the stringer 6 ℄	11-9	13,000	0.2	0.2	0.7
			a			0.2		0.2
			b					0.1
48	Same as crack 35		1.8" fwd of the stringer 6 ℄	11-9	13,000	0.1		0.1
			a			0.1		
46	Same as crack 35		1.8" aft of the stringer 6 ℄	11-9	13,000	0.1	0.9	0.9
			a				0.5	0.5
			b			0.1	0.2(b)	0.2
56	Same as crack 35		1.6" aft of the stringer 7 ℄	11-9	14,250	0.9		0.9
			a			0.3		0.3
			b			0.3		0.3
39	Same as crack 35		1.8" fwd of the stringer 11 ℄	11-10	12,750	0.2	0.3	0.3
			a			0.2	0.3	0.3
81	Same as crack 35	213.7L	2.6" aft of the stringer 11 ℄	11-10	14,500	0.6		0.6
						0.1		0.1
						0.2		0.2

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-11

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered (Inches)	(d) Length After 14,500 Cycles	Length After Failure at 57.3% Limit Load
42	The crack originated in the edge of a fastener hole common to the door doubler and skin	203.6L	2.7" fwd of the stringer 2 ℄ b	11-16	12,750	0.1 0.1		0.1 0.1
6	The crack originated in the edge of a fastener hole common to the door doubler and skin. Was one of a row adjacent to the access door attachment fasteners	199.5L	0.5" fwd of the stringer 5 ℄ a		2,000	0.1 0.1	0.2 0.2	0.2 0.2
78	Same as crack 42	179.2L	2.0" fwd of the stringer 6 ℄ a	11-13	14,500	0.2 0.2		0.2 0.2
79	The crack originated in the edge of a fastener hole common to the door doubler, skin and stringer leg	176.4L	2.0" aft of the stringer 5 ℄ a b	11-13	14,500	0.7 0.2 0.3		0.7 0.2 0.3
43	Same as crack 42	174L	The stringer 2 ℄ a	11-15	12,750	0.1 0.1		0.1 0.1
44	Same as crack 42	174L	The stringer 5 ℄ a b	11-13	12,750	0.2 0.2	0.7 0.1 0.2	0.8 0.1 0.3
36	The crack originated in the edge of a fastener hole common to a stringer leg and the skin	151L	1.8" aft of the stringer 5 ℄ a b	11-17	12,000	0.1 0.1	0.7 0.2	0.7 0.1 0.2
37	Same as crack 36	152.6L	1.8" aft of the stringer 4 ℄ b	11-17	12,250	0.2 0.2		0.2 0.2

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-11

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered (Inches)	(d) Length After 14,500 Cycles	Length After Failure at 57.3% Limit Load
25	The crack originated in the edge of a fastener hole common to the spar cap and skin	140.6L	1.2" aft of the fwd edge of panel #1	11-27	10,250	0.3 0.3	0.3 0.3	0.4 0.4
			[b]					
77	Same as crack 36	140.5L	2.0" fwd of the stringer 3 ℄	11-26	14,500	0.1 0.1		0.1 0.1
			[a]					
75	Same as crack 36	91.2L	1.8" fwd of the stringer 6 ℄	11-18	14,500	0.2 0.2		0.2 0.2
			[b]					
76	Same as crack 36	88.2L	1.8" fwd of the stringer 6 ℄	11-18	14,500	0.2 0.2		0.2 0.2
			[a]					
66	Same as crack 36	63.5L	1.8" fwd of the stringer 7 ℄	11-25	14,500	0.2 0.2		0.2 0.2
			[a]					
23	The crack originated in the edge of a fastener hole common to the rib, stringer leg and skin	60.5L	2.6" fwd of the stringer 2 ℄	11-14	10,250	0.3	0.7 0.1	0.8 0.2
			[a] [b]			0.3(c)		0.3
24	The crack originated in the edge of a fastener hole common to the rib and the skin	60.5L	2.6" aft of the stringer 1 ℄	11-14	10,250	0.3 0.3	0.4 0.4	0.4 0.4
			[b]					
62	Same as crack 24	60.5L	2.4" fwd of the stringer 8 ℄	11-25	14,500	0.2 0.2		0.2 0.2
			[b]					
63	Same as crack 23	59.4L	1.8" fwd of the stringer 3 ℄		14,500	0.9 0.1 0.4		0.9 0.1 0.4
			[a] [b]					

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-11

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered (Inches)	(d) Length After 14,500 Cycles	Length After Failure at 57.3% Limit Load
64	Same as crack 36	57.2L	1.8" fwd of the stringer 3 ℄		14,500	0.3		0.3
			b			0.3		0.3
65	Same as crack 36	56.4L	1.8" fwd of the stringer 4 ℄		14,500	0.9		0.9
			a			0.3		0.3
			b			0.2		0.2
61	Same as crack 36	48.7L	1.7" aft of the stringer 5 ℄		14,500	0.2		0.2
			b			0.2		0.2
52	Same as crack 36	47L	1.8" fwd of the stringer 2 ℄		13,000	0.1		0.1
			b			0.1		0.1
50	Same as crack 6	5.7L	0.6" fwd of the stringer 5 ℄	11-19	13,000	0.6	0.7	0.8
			a			0.2	0.2	0.4(b)
			b			0.2		0.2
30	Same as crack 6	10.5R	0.6" fwd of the stringer 5 ℄	11-19	10,750	0.3	0.4	0.4
			a			0.3	0.4	0.4
29	The crack originated in the edge of a fastener hole adjacent to an external skin repair doubler	52.3R	1.8" aft of the stringer 5 ℄	11-20	10,750	0.2		0.2
			b			0.2		0.2
45	Same as crack 29	52.3R	1.8" fwd of the stringer 5 ℄	11-20	12,750	0.1	0.2	0.2
			a			0.1	0.2	0.2
51	Same as crack 29	52.3R	1.8" fwd of the stringer 2 ℄	11-20	13,000	0.1	0.6	0.6
			a			0.1	0.2	0.2
			b				0.1	0.1

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-11

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered (Inches)	(d) Length After 14,500 Cycles	Length After Failure at 57.3% Limit Load
60	Same as crack 29	52.3R	1.7" aft of the stringer 2 ℄	11-20	14,500	0.1		0.1
			a			0.1		0.1
68	Same as crack 36	57.1R	1.8" aft of the stringer 2 ℄	11-20	14,500	0.8		0.8
			a			0.3		0.3
			b			0.2		0.2
69	Same as crack 36	59.3R	1.8" fwd of the stringer 2 ℄	11-20	14,500	0.1		0.1
			a			0.1		0.1
20	Same as crack 24	60.5R	3" fwd of the stringer 8 ℄	11-21	10,250	0.2	0.3	0.3
			b			0.2	0.3	0.3
59	Same as crack 24	60.5R	2.6" fwd of the stringer 11 ℄	11-22	14,500	0.7		0.7
			a			0.1		0.1
			b			0.3		0.3
4	Same as crack 25	60.5R	1.4" fwd from aft edge of Panel #4	11-22	800	0.1	0.6	0.6
			a			0.1	0.6	0.6
21	Same as crack 24	61.5R	4" fwd of the stringer 8 ℄	11-21	10,250	0.1		0.1
			b			0.1		0.1
74	Same as crack 36	63.5R	1.8" fwd of the stringer 2 ℄	11-23	14,500	0.1		0.1
			b			0.1		0.1
73	Same as crack 36	67.5R	1.8" fwd of the stringer 2 ℄	11-23	14,500	0.3		0.3
			b			0.3		0.3
72	Same as crack 36	69.5R	1.8" fwd of the stringer 2 ℄	11-23	14,500	0.2		0.2
			b			0.2		0.2

NOTE: Footnotes found in Table III of the General Section, Part G.

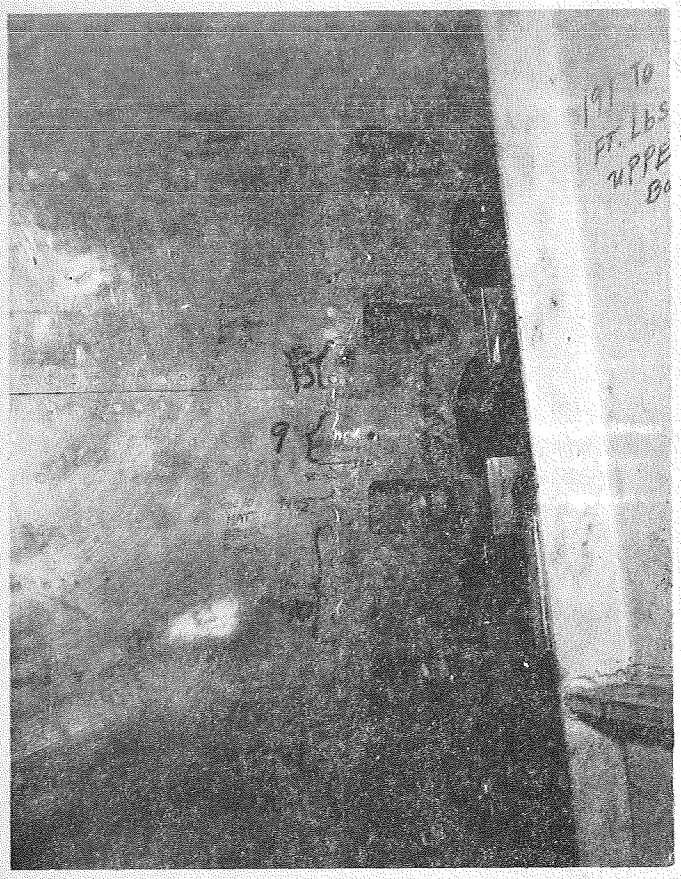
TABLE II-11

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 11

UPPER SURFACE SKIN, (a)

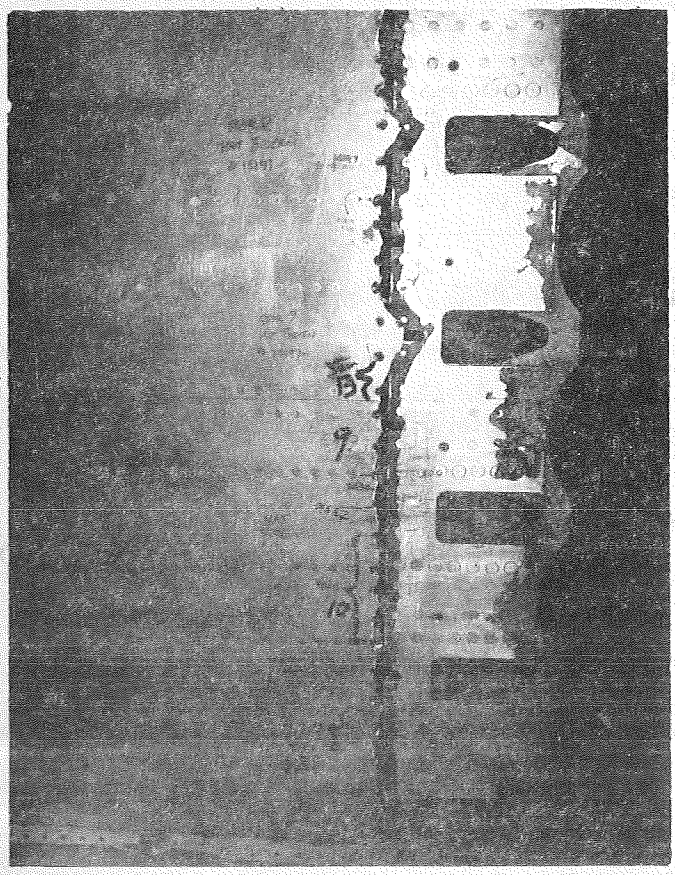
Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered (Inches)	(d) Length After 14,500 Cycles	Length After Failure at 57.3% Limit Load
67	Same as crack 36	69.3R	1.8" fwd of the stringer 6 ℄	11-21	14,500	0.2		0.2
			b			0.2		
55	Same as crack 36	71R	1.9" fwd of the stringer 10 ℄	11-22	14,500	0.1	0.2	0.2
			a			0.1	0.2	0.2
70	Same as crack 36	73.5R	1.8" fwd of the stringer 2 ℄	11-23	14,500	0.2		0.2
			b			0.2		0.2
71	Same as crack 36	76.3R	1.8" fwd of the stringer 2 ℄	11-23	14,500	0.3		0.3
			a			0.3		0.3
19	Same as crack 25	101.5R	1.2" aft of the fwd edge of Panel #1	11-28	10,250	0.2		0.2
			b			0.2		0.2
18	Same as crack 25	139.3R	1.2" aft of the fwd edge of Panel #1		10,250	0.2		0.2
			a			0.2		0.2
27	Same as crack 25	178.5R	0.5" aft of the fwd edge of Panel #1	11-7	10,750	0.1		0.1
			a			0.1(b)		0.1
28	Same as crack 36	173.6R	The stringer 5 ℄		10,750	0.1		0.1
			b			0.1		0.1
58	Same as crack 6	199.5R	0.5" fwd of the stringer 5 ℄	11-8	14,500	0.3		0.3
			a			0.3		0.3(b)

NOTE: Footnotes found in Table III of the General Section, Part G.



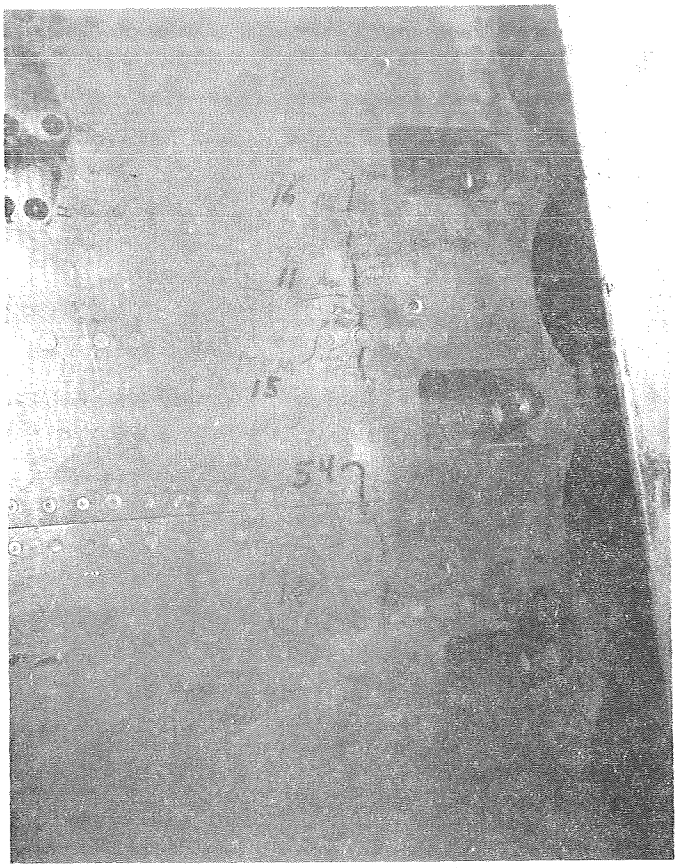
RK 0568-21

Figure 11-1
Cracks 9, 10, 12 and 13 at WS 213.5R after the crack propagation test of 14,500 cycles.



RK 0611-2

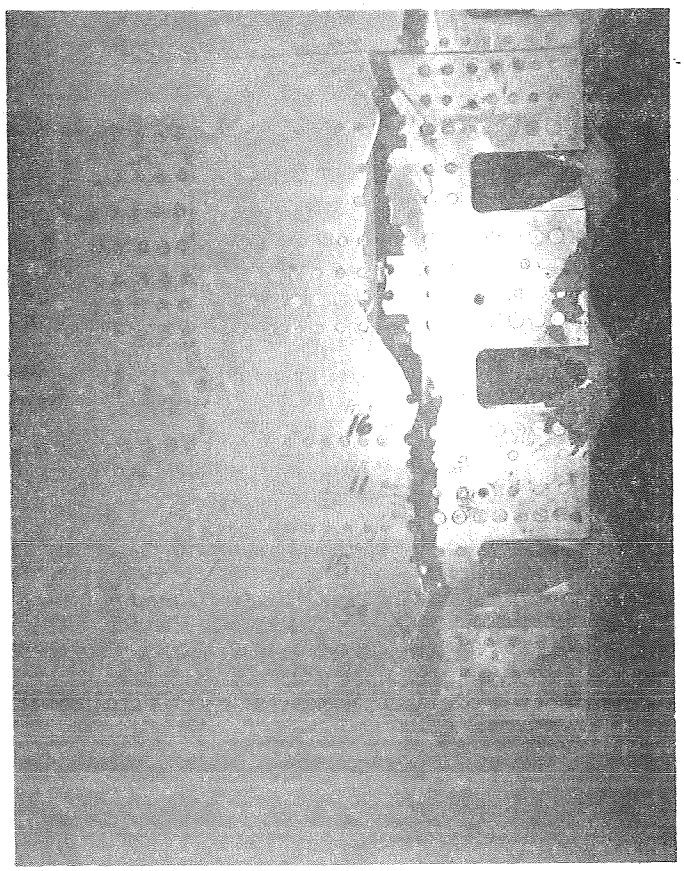
Figure 11-2
Chordwise failure occurred along cracks 9, 10, 12 and 13 at WS 213.5R at 57.3% limit load.



RK 0568-20

Figure 11-3

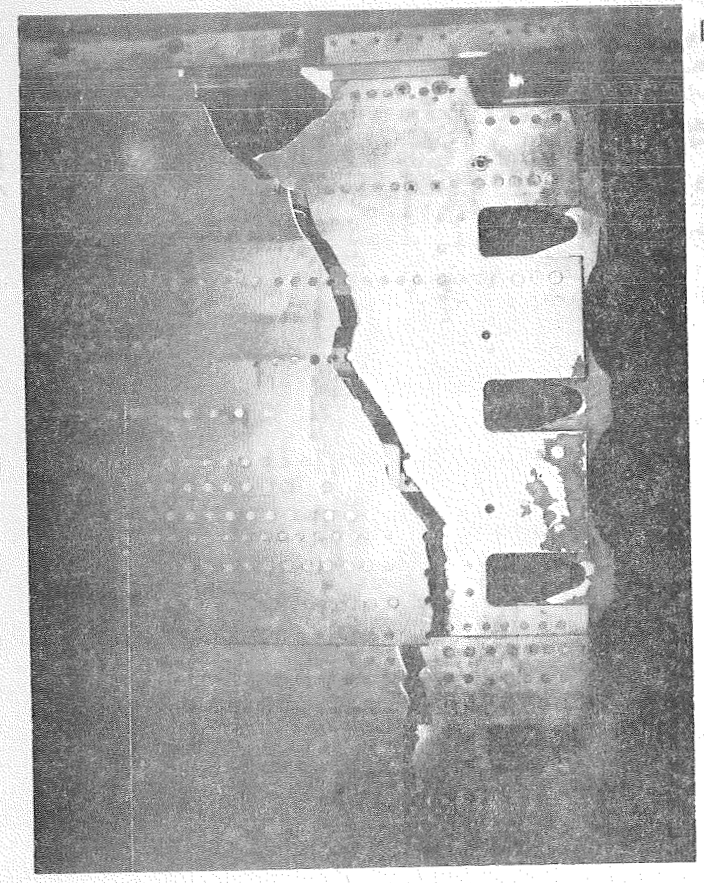
Cracks 11, 14, 15, 16, 54 & 57 at WS 213.5R after the crack propagation test of 14,500 cycles.



RK 0611-4

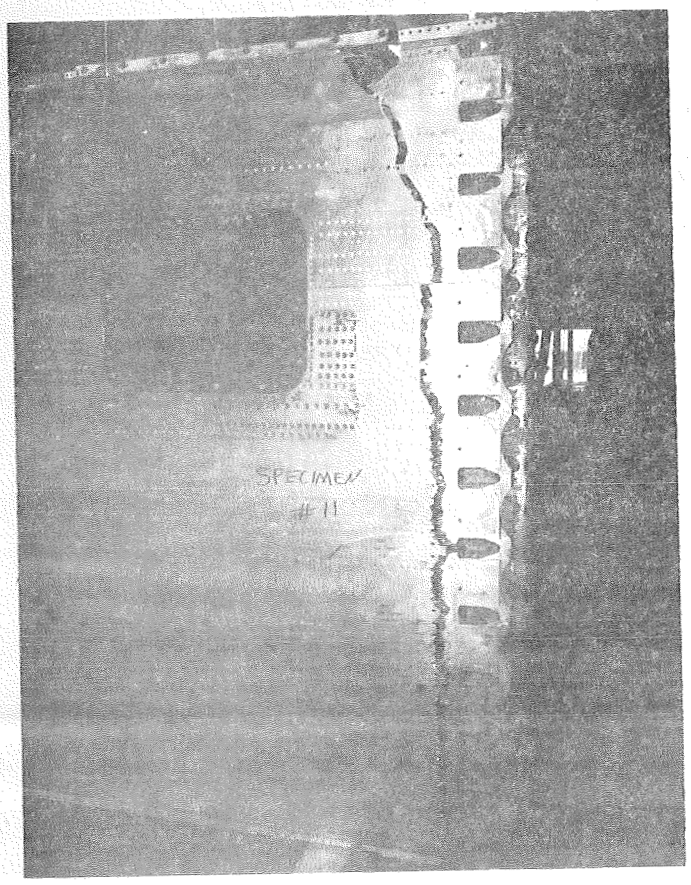
Figure 11-4

Chordwise failure along cracks 11, 14, 15, 16, 54 and 57 at WS 213.5R after 57.3% limit load.



RK 0611-3

Figure 11-5
Fwd end of the final chordwise failure after 57.3% limit load.



RK 0611-1

Figure 11-6
Overall view of the final chordwise failure.

RK 0568-19

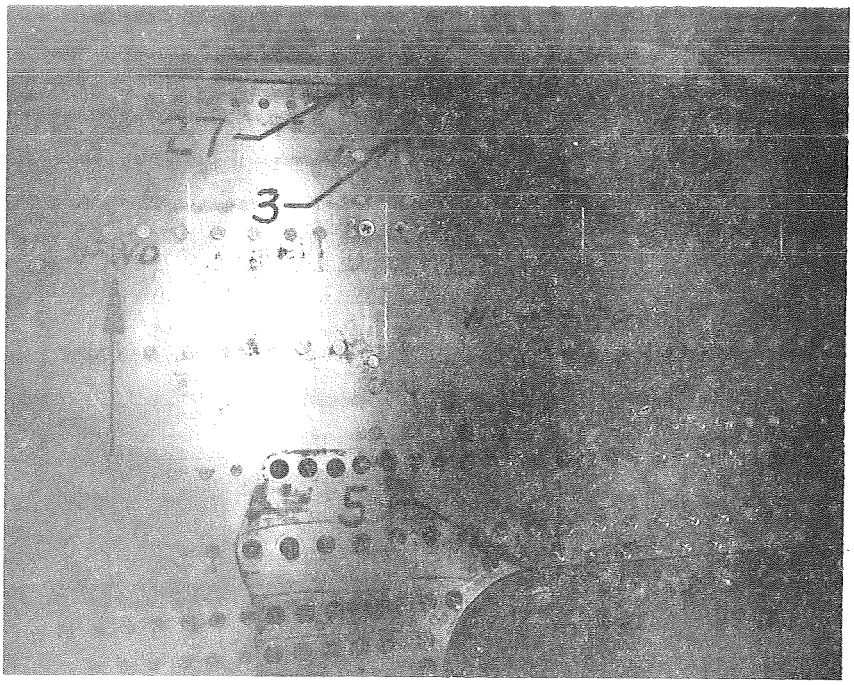


Figure 11-7 Locations of cracks 3, 5 & 27.

RK 0611-7

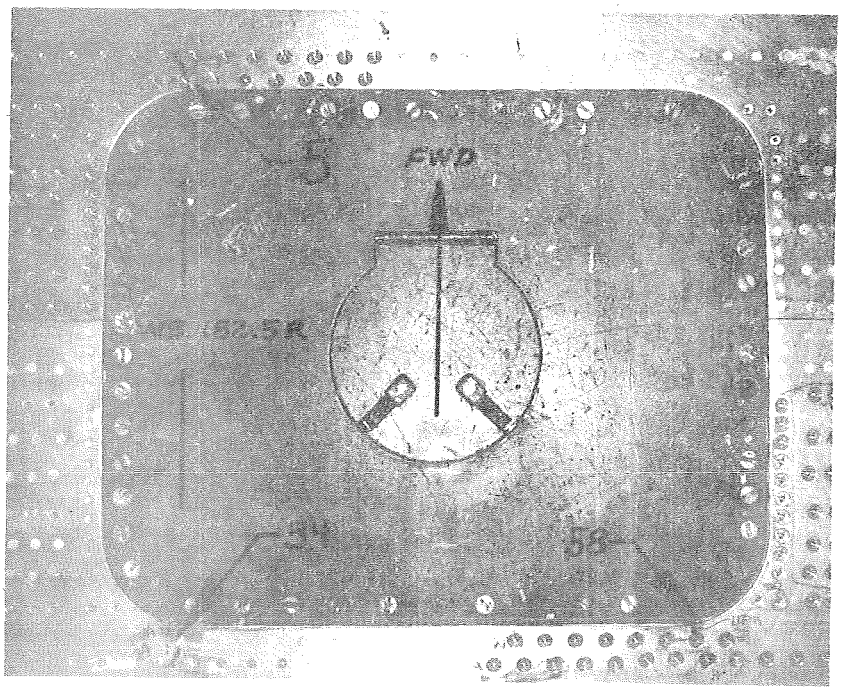
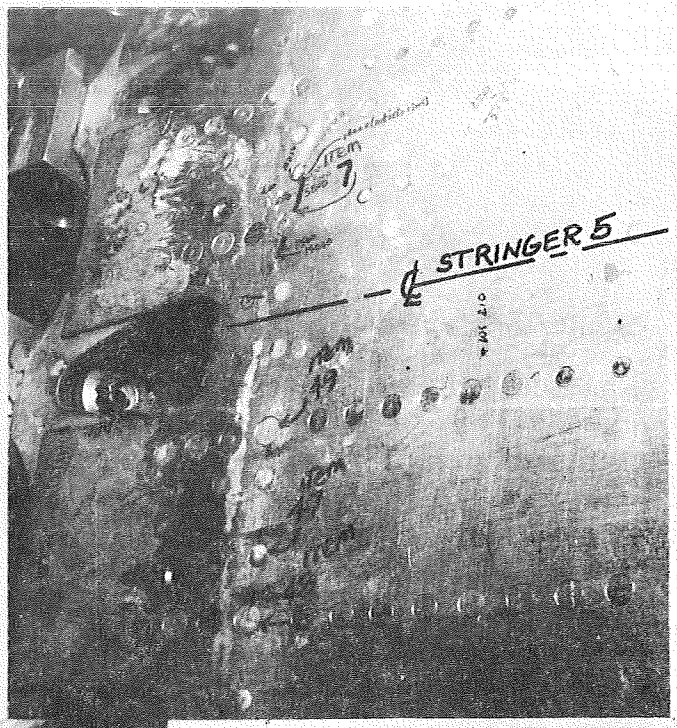


Figure 11-8 Locations of cracks 5, 34 & 58.

RK 0568-4



RK 0568-3

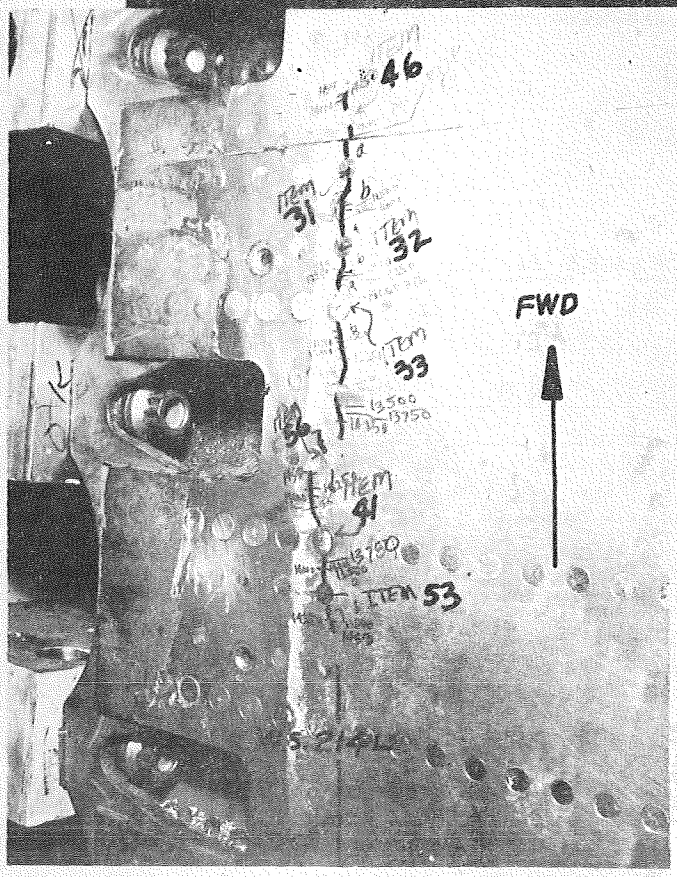
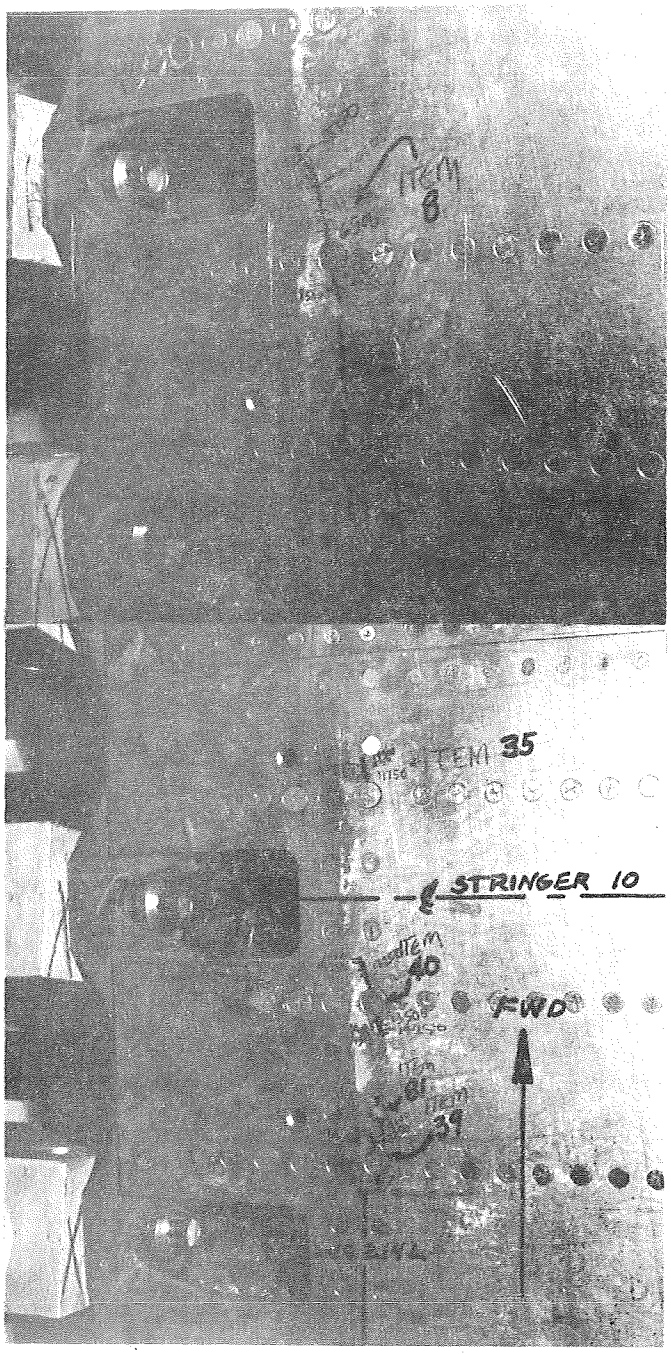


Figure 11-9

Cracking in the area of WS 214L as it appeared after completion of the crack propagation test of 14,500 cycles. Cracks 7, 31, 32, 33, 41, 46, 47, 48, 49, 53 and 56 are shown.

RK 0568-6



RK 0568-5

Figure 11-10 Cracking in the area of WS 214L as it appeared after completion of the crack propagation test of 14,500 cycles. Cracks 8, 35, 39, 40 & 81 are shown.

RK 0568-26

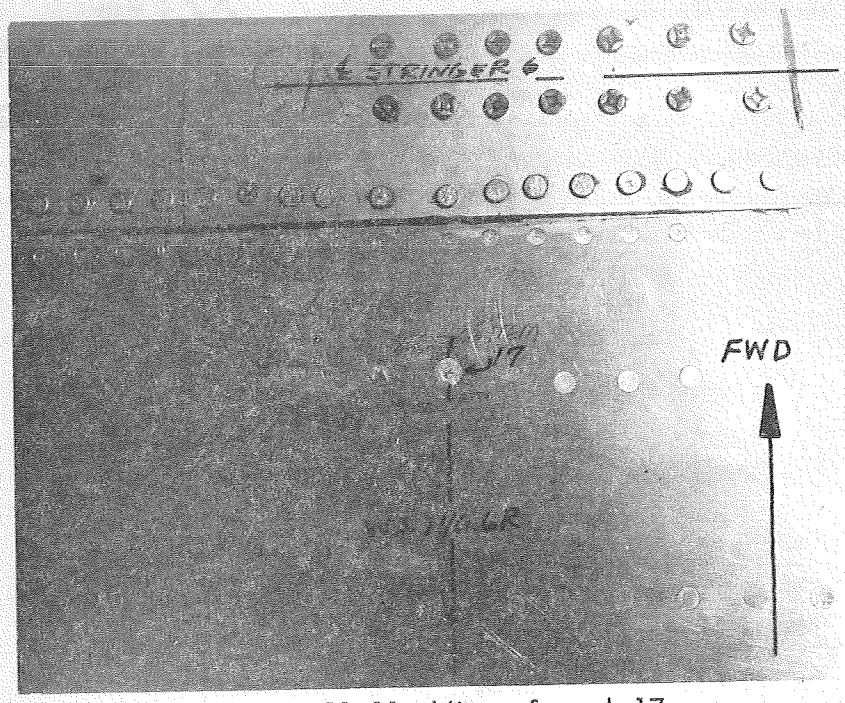


Figure 11-11 View of crack 17.

RK 0568-12

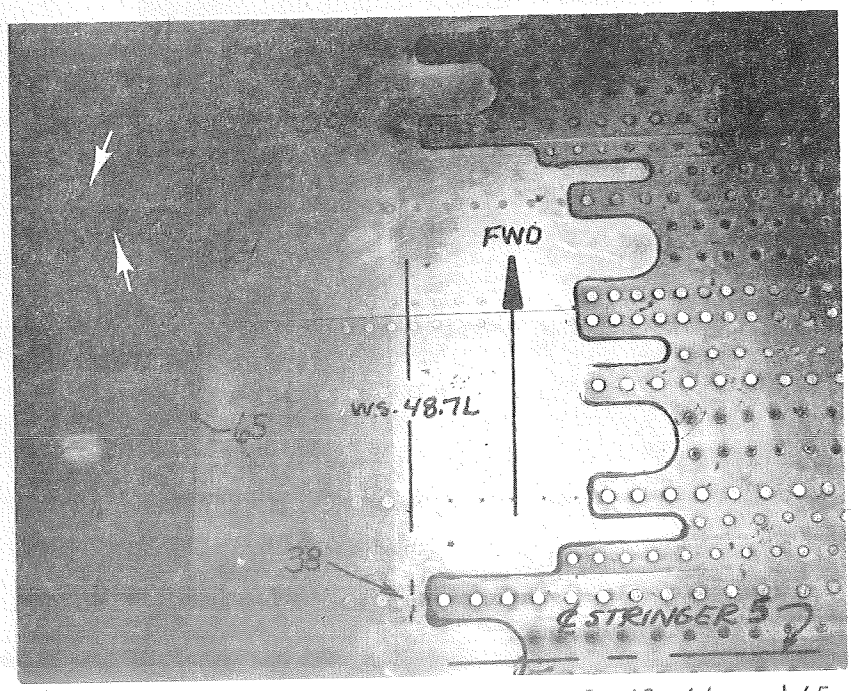


Figure 11-12 View of cracks 22, 38, 52, 63, 64, and 65.

RK 0568-17

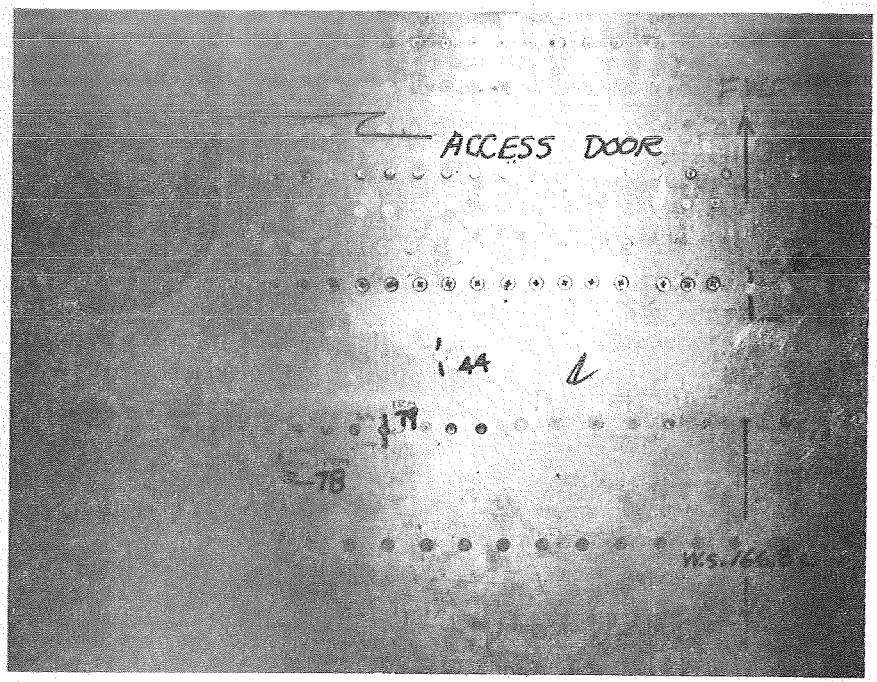


Figure 11-13 Cracks 44, 78, 79 & 80 adjacent to left access door.

RK 0568-11

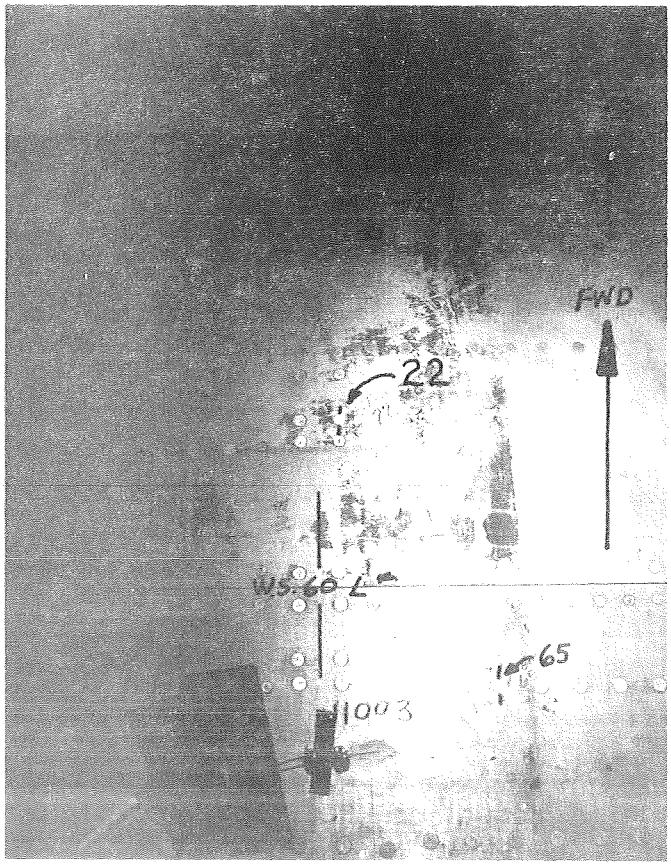


Figure 11-14 Cracks 22, 23, 24 & 65 are located in the WS 60 rib area.

RK 0568-2

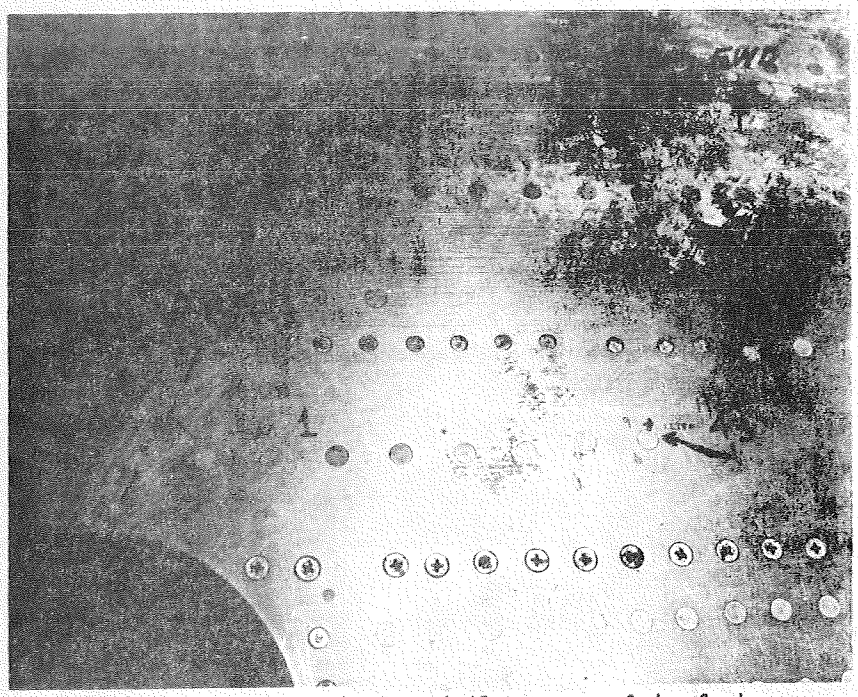


Figure 11-15 Cracks 1 and 43 in area of the fwd inboard corner of the left access door.

RK 0568-1

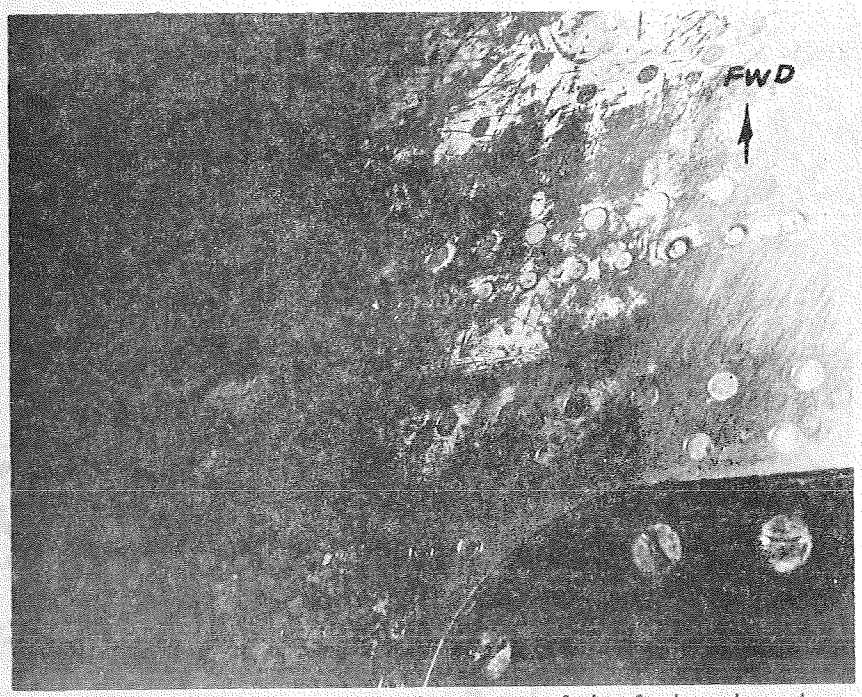


Figure 11-16 Crack 42 in area of the fwd outboard corner of the left access door.

RK 0568-18

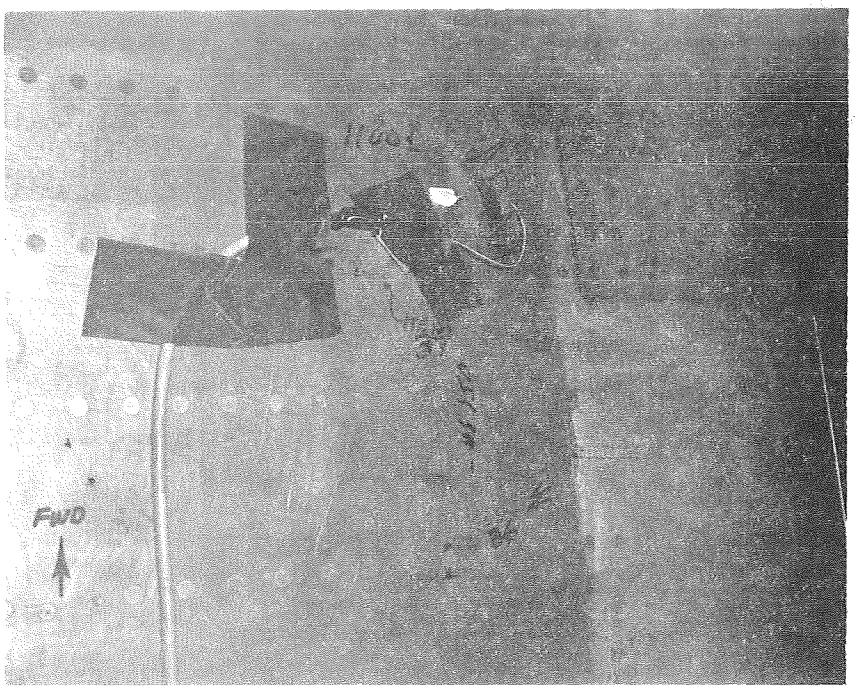


Figure 11-17 View of cracks 36 and 37.

RK 0568-13

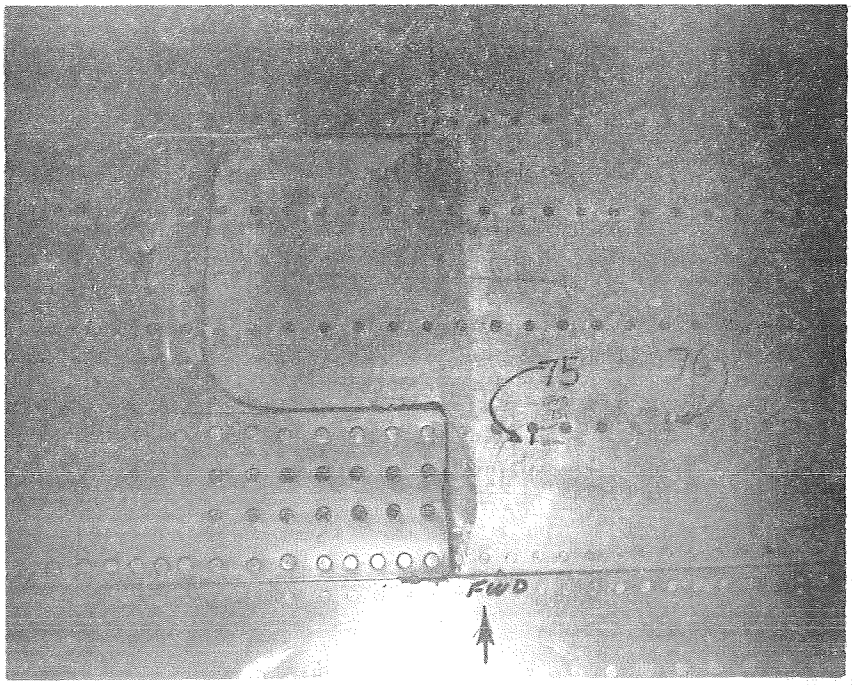


Figure 11-18 View of cracks 75 and 76.

RK 0568-9

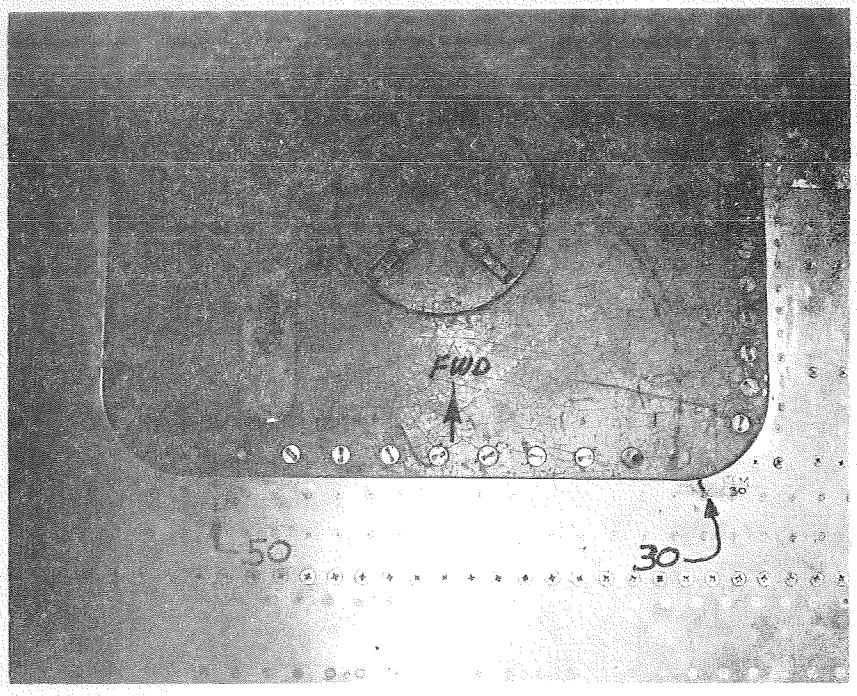


Figure 11-19 View of cracks 30 and 50.

RK 0568-8

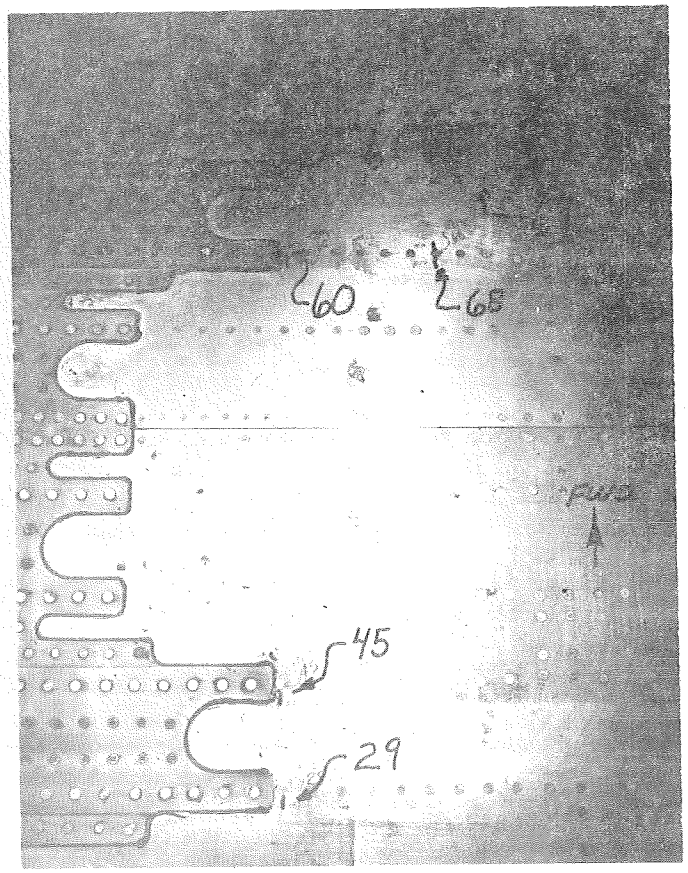


Figure 11-20 View of cracks 29, 45, 60, 61, 68 and 69.

RK 0568-7

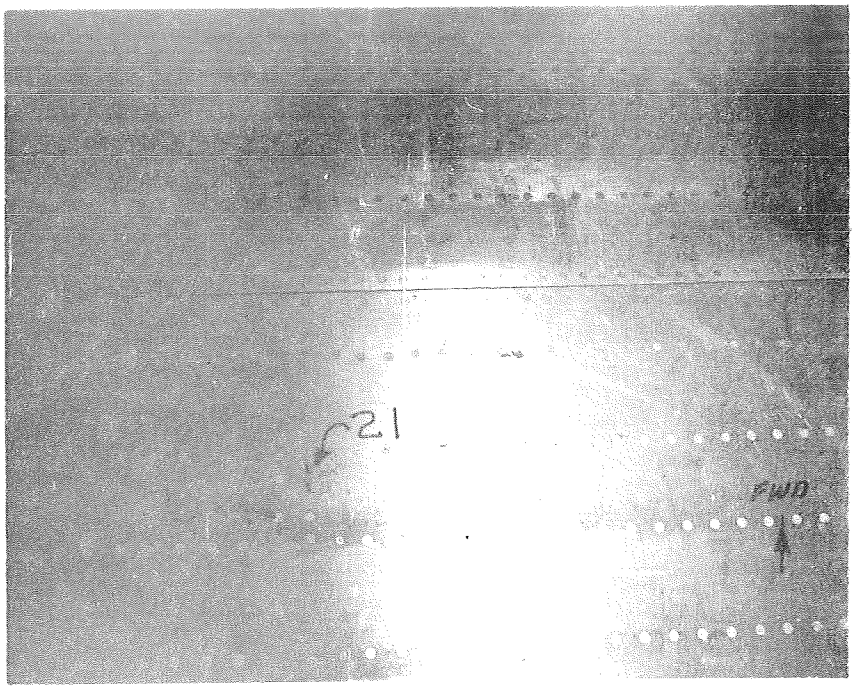


Figure 11-21 View of cracks 20, 21 and 67.

RK 0568-10

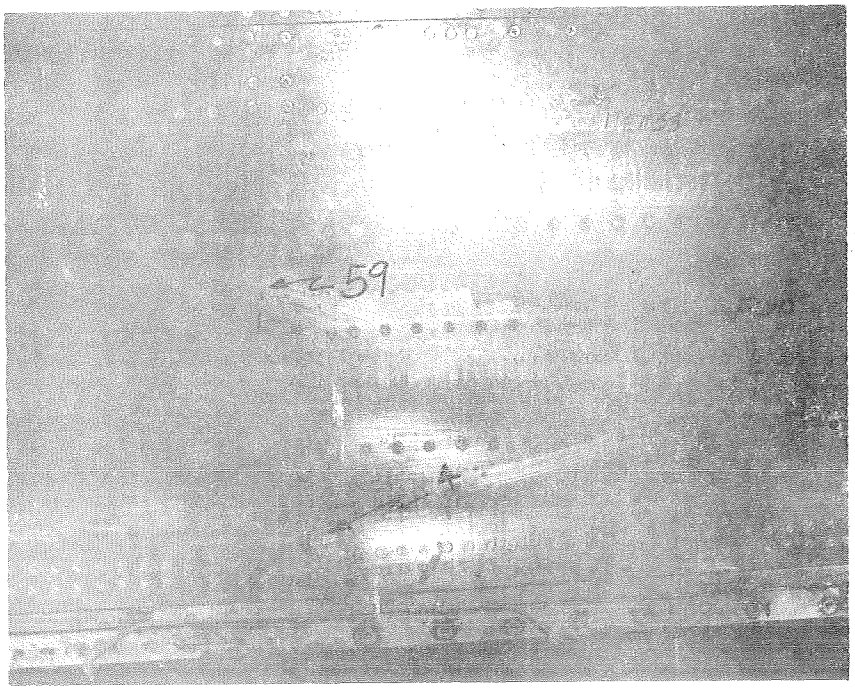


Figure 11-22 View of cracks 4, 55 and 59.

RK 0568-25

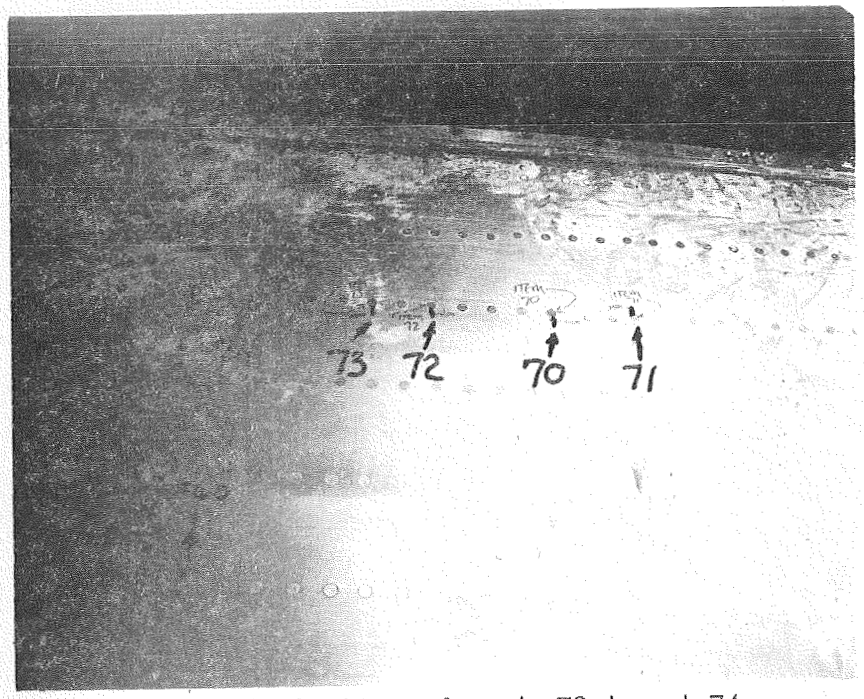


Figure 11-23 View of cracks 70 through 74.

RK 0568-23

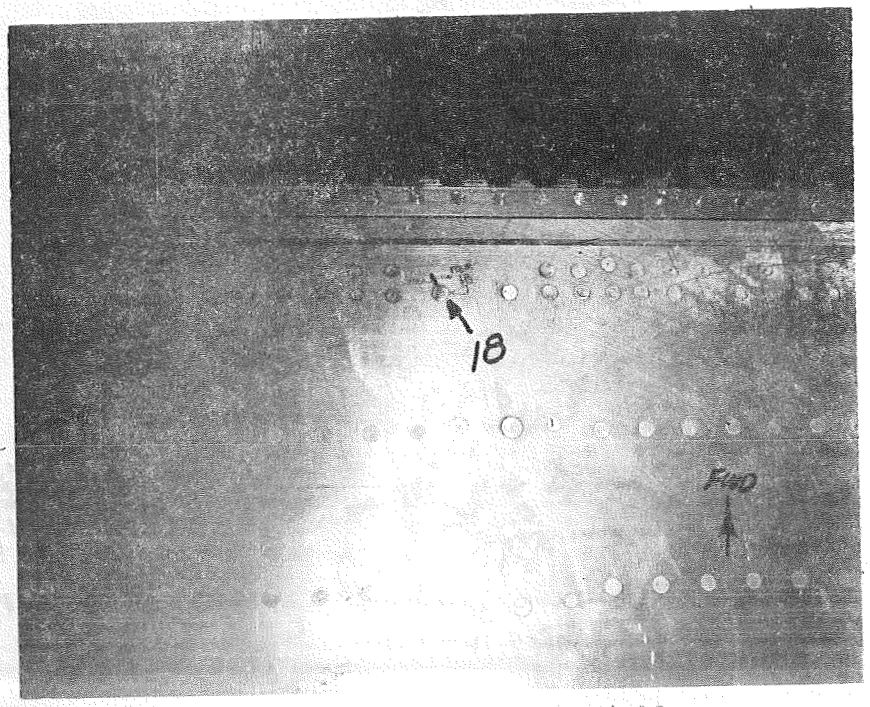


Figure 11-24 View of crack 18.

RK 0568-14

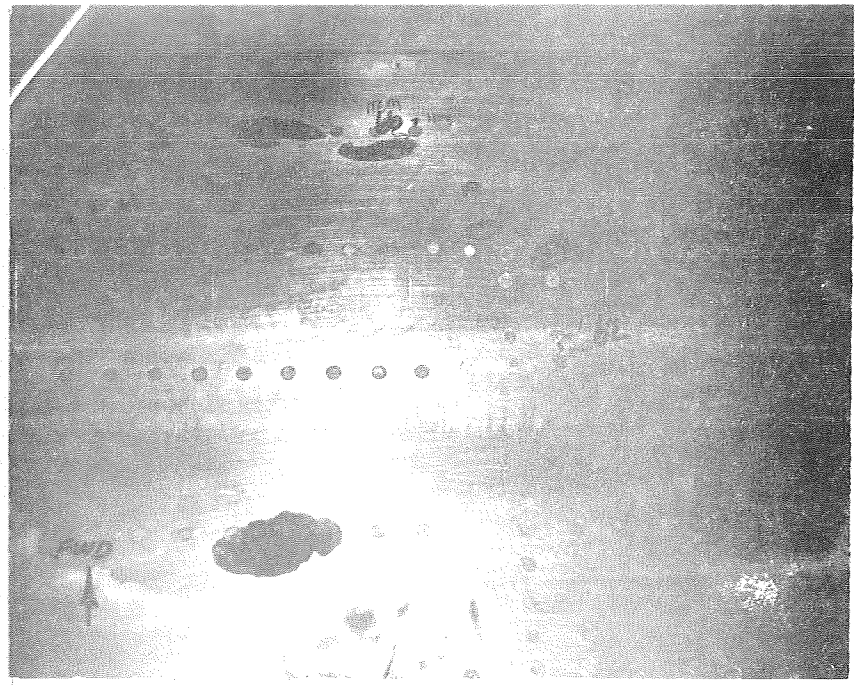


Figure 11-25 View of cracks 62 and 66.

RK 0568-15

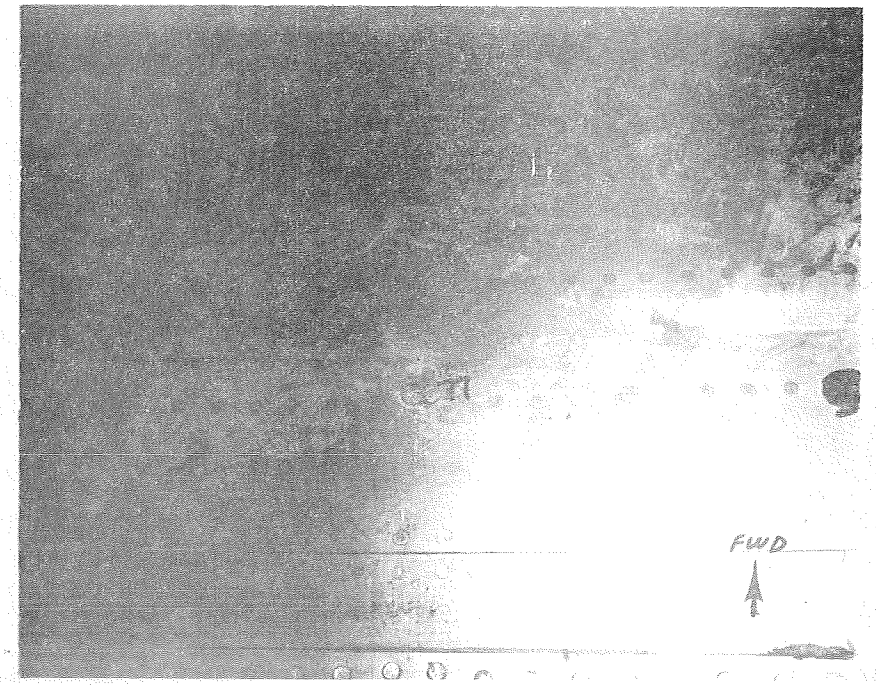


Figure 11-26 View of crack 77.

RK 0568-16

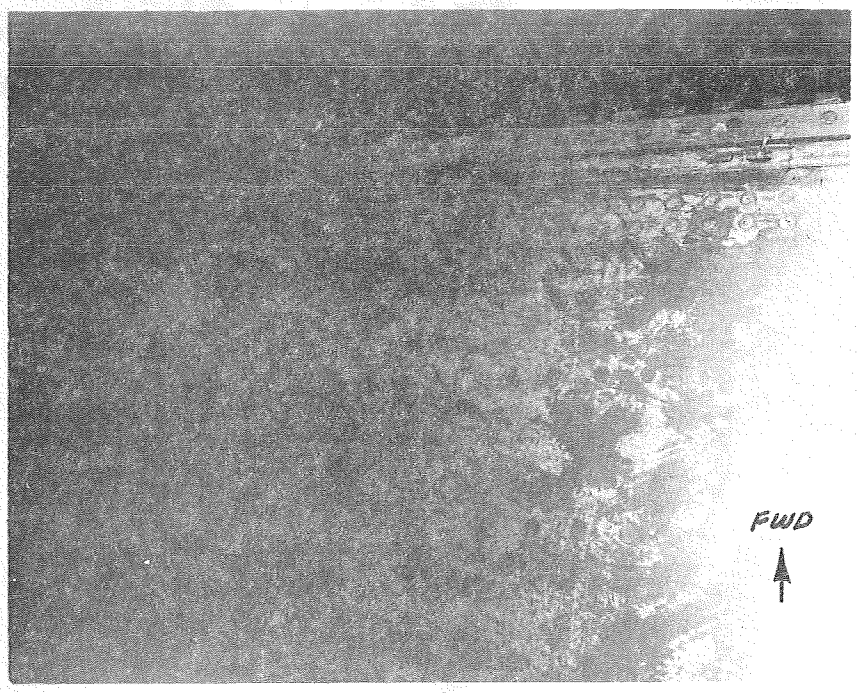


Figure 11-27 View of crack 25.

RK 0568-24

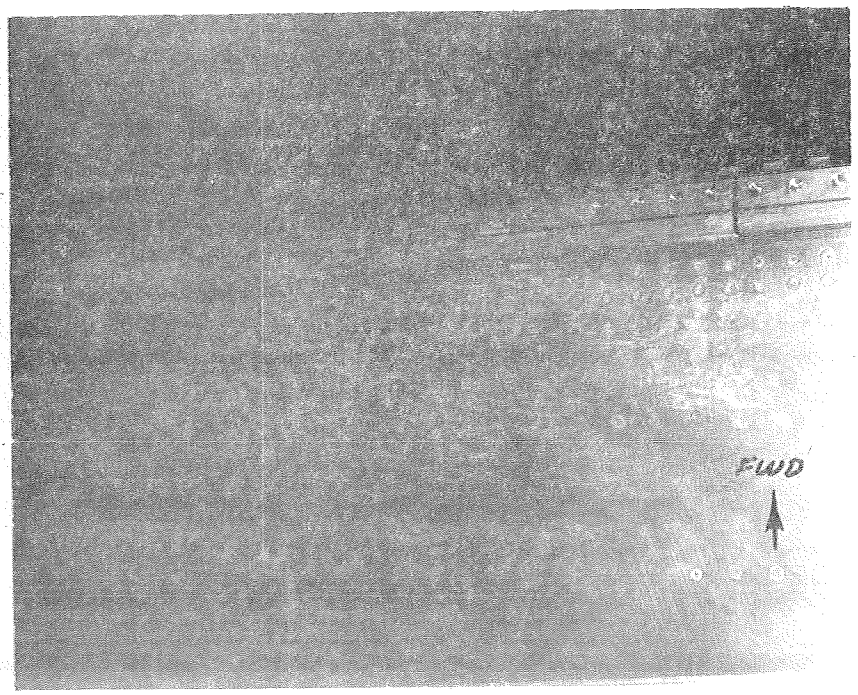


Figure 11-28 View of crack 19.

PART 12

Specimen 12

CRACK PROPAGATION & RESIDUAL STRENGTH DOWNBENDING TEST RESULTS

TEST RESULTS

A total of ten initial fatigue cracks were located in the upper surface skin. The longest cracks, numbers 4 and 5, were each 0.6 inches in length and were located in the doubler runout area at W.S. 105L. The maximum cyclic load for phase 1 of the crack propagation test was 50 percent limit load.

As shown in Table I-12, after 1000 cycles at 50% Limit Load, nine additional cracks appeared, and most of the ten initial cracks had grown perceptibly. While the crack growth history may be traced explicitly in Table I-12, the extent and rapidity of crack progression during subsequent cyclic testing of Specimen #12 merits special attention, as described in the following paragraphs.

Individual cracks at 1500 cycles grew or joined others so that, at 2000 cycles, crack 3 was 9.5 inches long at WS 105L. This 9.5-inch crack then grew to a length of 19.4 inches by 3000 cycles. Also, small cracks at WS 135L had grown and joined to form one 6.5-inch crack (number 8).

At 2870 cycles, Stringer #5 was discovered to be broken at WS 105L. Cyclic load level was reduced to 40% Limit Load after 4034 cycles. At that time, a total of 30 individually identified cracks had been detected. The large crack length at WS 105L measured 19.7 inches and the hat section of Stringer #4 was discovered to be cracked.

At 7036 cycles, crack 30 in the same vicinity of existing damage (WS 105, 107) extended from 3.0 inches to 6.3 inches in length. This crack continued to grow to 19.1 inches at 10,500 cycles (see Figures 12-1 and 12-2). At this point cycling was interrupted and an access door was removed to permit interior inspection of the hat section stringers at WS 105. In addition to the cracked Stringers #4 and #5 externally observed previously, Stringers #3, #6, #7 and #8 were also found to be cracked; see Figures 12-3 and 12-4.

The damage found was so extensive that cyclic testing was terminated, and static residual strength test to destruction was initiated. At inspection after 50% Limit Load (Condition D-2), a 6.9-inch crack, number 32, at WS 107L had grown to 10.5 inches. More sonic reports were heard at 53% Limit Load; inspection showed that the previous 10.5-inch crack had grown to 17.0 inches, and other cracks had grown lesser amounts. At this stage, one continuous crack extended over both interior skin panels and across approximately 80% of the width of the forward panel (out of a total of four upper surface skin panels). Total chordwise surface failure occurred at 56% Limit Load at WS 105L; see Figure 12-5. Crack propagation behavior around the fuel filler access opening is shown in Figures 12-6, 12-7, 12-12, and 12-3. Other cracks, mostly through end fasteners common to skins and reinforcing doublers, appear in Figures 12-8, 12-9, 12-10, 12-11, and 12-14. Table I-12 gives a summary of damage that occurred to the specimen.

TABLE I-12

SUMMARY OF CRACK LENGTHS, SPECIMEN 12

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Downbending 1, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d)								
					0	500	1000	1500	2000	2500	3000	3500	4000
32	The crack originated in the edge of a fastener hole common to the skin and stringer flange	107L	2.8" aft of the stringer 3 ℄ a b	12-1									
3	The crack originated in the edge of a fastener hole common to the skin and fuel filler hole doubler runout	105L	2.8" fwd of the stringer 5 ℄ a b	12-1	0.1 0.1		0.2 0.2	0.6 0.1 0.2	9.5 1.8 7.4		19.4 7.7 11.4(b)	19.7 8.0	19.7 8.0 11.4
18	Same as crack 3		0.5" fwd of the stringer 5 ℄ a b	12-1			0.4 0.1 0.1	0.6 0.2 0.2					
6	Same as crack 3		0.5" aft of the stringer 5 ℄ a b	12-1	0.1 0.1 -		0.6 0.2 0.1	0.9 0.3 0.3					
4	Same as crack 3		2.8" aft of the stringer 5 ℄ a b	12-1	0.6 0.2 0.1	0.7 0.3 0.2	0.8 0.4 0.2	2.7 1.1 1.3					
5	Same as crack 3	105L	2.8" fwd of the stringer 6 ℄ a b	12-1	0.6 0.1 0.2	0.8 0.2 0.3	0.9 0.3 0.3						
30	Same as crack 32	107L	2.8" aft of the stringer 6 ℄ a b	12-2									

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-12 (Cont.)

SUMMARY OF CRACK LENGTHS, SPECIMEN 12

UPPER SURFACE SKIN, (a)

Crack No.	Downbending 1, Phase 2 Cyclic Test, R = 0.1 Max. Load = 30% Limit Load — Cumulative Number of Cycles (d)									Downbending 2 Residual Strength Test		
	4500	5000	6500	7000	7036	9034	9500	10,000	10,500	After Applying 50% Limit Load	After Applying 53% Limit Load	After Failure at 56% Limit Load
32									6.9	10.5	17.0	
a									6.1(c)	9.7	16.2	
b									0.5(b)		0.5	
3	19.7								19.7(1)		19.7	
a	8.0								8.0			
b	11.4								11.4			
18												Complete Chordwise Failure
6												
4												
5												
30	2.6		2.6	3.0	6.3	9.1	12.6	12.6	19.1	19.1	19.9	
a	0.3(b)										0.3	
b	2.0(c)		2.0	2.4	5.7	8.5	12.0	12.0	18.5	18.5	19.3	

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-12 (Cont.)

SUMMARY OF CRACK LENGTHS, SPECIMEN 12

UPPER SURFACE SKIN, (a)

Downbending 1, Phase 1
Cyclic Test, R = 0.1
Max. Load = 50% Limit Load

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Number of Cycles (d) 0	500	1000	1500	2000	2500	3000	3500	4000
1	The crack originated in the edge of a fastener hole common to the fuel filler fitting and the skin	120.5L	2.5" fwd of the stringer 5 ℄	12-6	0.3	1.2	1.7						1.7
			a	12-7		0.6	1.1(c)						1.1
			b		0.3(c)								0.3
2	Same as crack 1	120.5L	2.5" aft of the stringer 5 ℄	12-6	0.3	1.6							1.6
			a	12-7	0.3(c)								0.3
			b			1.0(c)							1.0
26	Same as crack 3	135L	2.7" aft of the stringer 4 ℄	12-8						0.2	0.2		
			a							0.2	0.2		
7	Same as crack 3		2.8" fwd of the stringer 5 ℄	12-8	0.1					0.1	0.3		
			a		0.1					0.1	0.3		
19	Same as crack 3		0.5" fwd of the stringer 5 ℄	12-8			0.1	0.6	0.6				
			a				-	0.1	0.1				
			b				0.1	0.2	0.2				
8	Same as crack 3		0.7" aft of the stringer 5 ℄	12-8	0.5	0.5	0.8	1.0	1.3	6.5	6.5	9.5	12.9
			a		0.1	0.1	0.2	0.3	0.5	2.2(c)	2.2	5.5(c)	5.5(c)
			b		0.1	0.1	0.3	0.4	0.5	4.0(c)		4.0	7.5(c)
23	Same as crack 3	135L	2.8" fwd of the stringer 6 ℄	12-8					0.6				
			a						0.2				
			b						0.1				
21	The crack originated in the edge of a fastener hole common to the door doubler and the skin	36L	0.5" fwd of the stringer 5 ℄	12-9					0.6	0.8	1.0	1.0	2.0
			a						0.2	0.3	0.4	0.4	0.8
			b						0.1	0.2	0.3	0.3	0.9(c)
22	Same as crack 21	36L	0.5" aft of the stringer 4 ℄	12-9					0.1	0.7	0.8	0.8	0.9
			a						0.1	0.2			0.2
			b							0.1	0.2	0.2	0.3

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-12 (Cont.)

SUMMARY OF CRACK LENGTHS, SPECIMEN 12

UPPER SURFACE SKIN, (a)

Crack No.	Downbending 1, Phase 2 Cyclic Test, R = 0.1 Max. Load = 30% Limit Load — Cumulative Number of Cycles (d)									Downbending 2 Residual Strength Test		
	4500	5000	6500	7000	7036	9034	9500	10,000	10,500	After Applying 50% Limit Load	After Applying 53% Limit Load	After Failure at 56% Limit Load
1	1.7								1.7			1.7
a	1.1								1.1			
b	0.3								0.3			
2	1.6								1.6			1.6
a	0.3								0.3			
b	1.0								1.0			
26												
7												
19												
8	12.9								12.9			12.9
a	5.5								5.5			
b	7.5								7.5			
23												
21	2.1								2.1			2.1
a	0.9								0.9			
b	0.9								0.9			
22	1.0								1.0			1.0
a	0.3								0.3			
b	0.3								0.3			

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-12 (Cont.)

SUMMARY OF CRACK LENGTHS, SPECIMEN 12

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Downbending 1, Phase 1 Cyclic Test, R = 0.1 Max. Load = 50% Limit Load — Number of Cycles (d)								
					0	500	1000	1500	2000	2500	3000	3500	4000
29	Same as crack 21	36L	0.5" aft of the stringer 2 ℄	12-10									0.2
			a										0.2
28	Same as crack 21	36L	0.5" fwd of the stringer 2 ℄	12-10						0.7	0.8	1.0	1.0
			a							0.2	0.3	0.5	
			b							0.2	0.2	0.3	0.3
27	Same as crack 21	36L	0.5" aft of the stringer 2 ℄	12-9						0.2	0.3	1.0	1.8
			a									0.4	0.5
			b							0.2	0.3	0.3	1.0
24	Same as crack 21	174.5L	2.0" fwd of the stringer 5 ℄	12-11						0.2	0.2	0.2	0.6
			a										0.1
			b							0.2	—	—	0.2
25	Same as crack 21	173.5L	The stringer 5 ℄	12-11						0.2	—	—	0.2
			b							0.2	—	—	0.2
9	Same as crack 1	120.5R	2.5" fwd of the stringer 5 ℄	12-12	0.1	—	1.1	1.6	—	—	—	—	1.6
			a				0.5	1.0(c)	—	—	—	—	1.0
			b		0.1		0.3(c)	—	—	—	—	—	0.3
10	Same as crack 1	120.5R	2.5" aft of the stringer 5 ℄		0.3	1.1	1.8	2.2	3.0	3.2	3.3	3.9	4.2
			a		0.3(c)								0.3
			b			0.5	1.2	1.6	2.4	2.6	2.7	3.3	3.6
11	Same as crack 3	105R	2.8" aft of the stringer 4 ℄				0.5	0.5	0.6	0.9	2.7	6.5	11.0
			a				0.1	0.1	0.2	0.3	0.7(c)		0.7
			b				0.2	—	0.2	0.3	1.7	5.5(c)	10.0
15	Same as crack 3		2.8" fwd of the stringer 5 ℄				0.5	0.5	0.8	0.9	0.9		
			a				0.1	0.1	0.2	0.3			
			b				0.1	0.1	0.3	0.4	0.4		
20	Same as crack 3		0.5" fwd of the stringer 5 ℄						0.2	0.2	0.3		
			a						0.2	0.2	0.3		
12	Same as crack 3		0.5" aft of the stringer 5 ℄				0.6	0.6	0.9	1.1	1.8		
			a				0.2	0.2	0.3	0.3	0.6		
			b				0.1	0.2	0.3	0.4	0.9		
14	Same as crack 3		2.8" fwd of the stringer 6 ℄				0.1	0.1	0.2	0.6	0.8	1.0	
			a				0.1	0.1	0.2	0.2	0.3	0.4	
			b							0.2	0.2	0.4	

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-12 (Cont.)

SUMMARY OF CRACK LENGTHS, SPECIMEN 12

UPPER SURFACE SKIN, (a)

Crack No.	Downbending 1, Phase 2 Cyclic Test, R = 0.1 Max. Load = 30% Limit Load — Cumulative Number of Cycles (d)									Downbending 2 Residual Strength Test		
	4500	5000	6500	7000	7036	9034	9500	10,000	10,500	After Applying 50% Limit Load	After Applying 53% Limit Load	After Failure at 56% Limit Load
29	0.2										0.2	
a	0.2										0.2	
28	1.0						1.0	1.1			1.1	2.2
a	0.3						0.3	0.5			0.5	0.6
b	0.3										0.3	1.3
27	1.8								1.8			1.8
a	0.5								0.5			
b	1.0								1.0			
24	0.6	0.7							0.7			0.7
a	0.1								0.1			
b	0.2	0.3							0.3			
25	0.2								0.2			0.2
a	0.2								0.2			
9	1.6								1.6			1.6
a	1.0								1.0			
b	0.3								0.3			
10	4.2								4.2			4.2
a	0.3								0.3			
b	3.6								3.6			
11	11.1								11.1			11.1
a	0.7								0.7			
b	10.1								10.1			
15												
20												
12												
14												

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-12 (Cont.)

SUMMARY OF CRACK LENGTHS, SPECIMEN 12

UPPER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Downbending 1, Phase 1 Cyclic Test, R = 0.1, Max. Load = 50% Limit Load Number of Cycles (d) 0 / 500 / 1000 / 1500 / 2000 / 2500 / 3000 / 3500 / 4000
15	Same as crack 3	104R	2.8" aft of the stringer 5 ℄		
			a		0.1 — 0.1 — 0.1 — 0.3 — 1.1
			b		0.1 — 0.1 — 0.1 — 0.3 — 0.7(c) 0.2
16	Same as crack 21	35R	0.5" fwd of the stringer 5 ℄		
			a		0.2 — 0.2 — 3.6 — 4.2 — 5.2 — 7.1
			b		0.2 — 0.2 — 0.8 1.1 1.2 1.5; 2.4 — 2.7 — 3.6 — 5.2(c)
17	Same as crack 21	35R	0.5" aft of the stringer 5 ℄		
			a		0.5 — 0.7 — 0.9
			b		0.1 — 0.2 — 0.3
31	Same as crack 21	35R	0.5" aft of the stringer 4 ℄		0.1 — 0.2 — 0.3

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-12 (Cont.)

SUMMARY OF CRACK LENGTHS, SPECIMEN 12

UPPER SURFACE SKIN, (a)

Crack No.	Downbending 1, Phase 2 Cyclic Test, R = 0.1 Max. Load = 30% Limit Load — Cumulative Number of Cycles (d)									Downbending 2 Residual Strength Test		
	4500	5000	6500	7000	7036	9034	9500	10,000	10,500	After Applying 50% Limit Load	After Applying 53% Limit Load	After Failure at 56% Limit Load
13	1.1								1.1			1.1
a	0.7								0.7			
b	0.2								0.2			
16	7.1										7.1	7.2
a	1.5										1.5	1.7
b	5.2											5.2
17												
31		0.2										0.2
a		0.2										0.2

NOTE: Footnotes found in Table III of the General Section, Part G.

RK 686-1

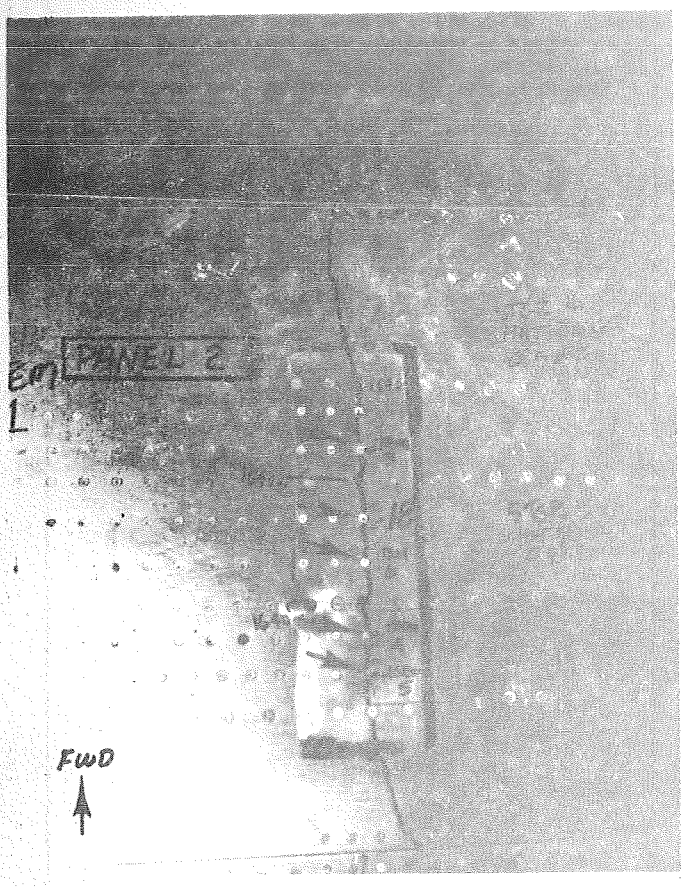


Figure 12-1

Area of the general chordwise failure, WS 105L, as it appeared after completing the propagation test of 10,500 cycles.

RK 686-4

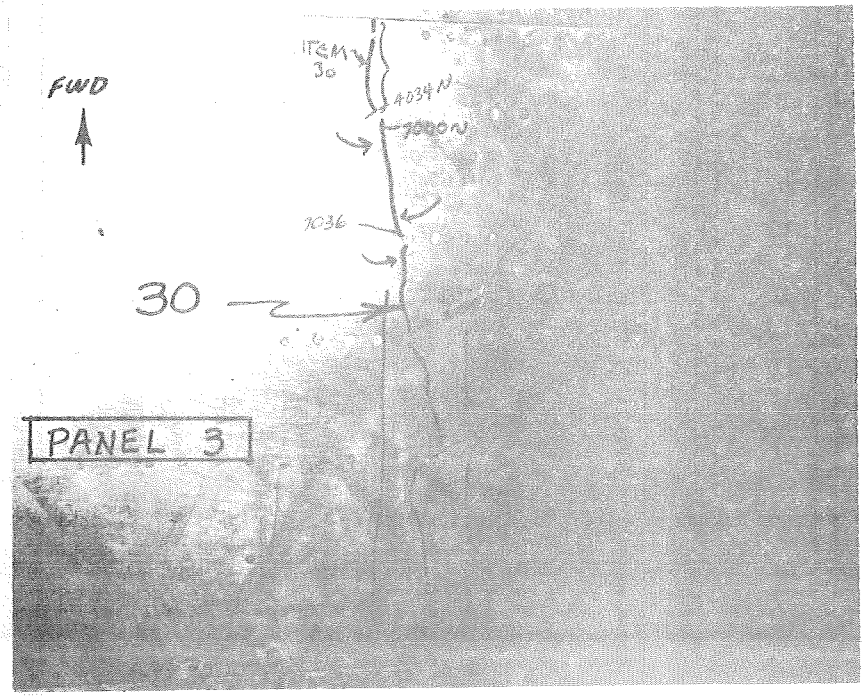


Figure 12-2

Area of the general chordwise failure, WS 105L, as it appeared after completing the propagation test of 10,500 cycles.

RK 686-8

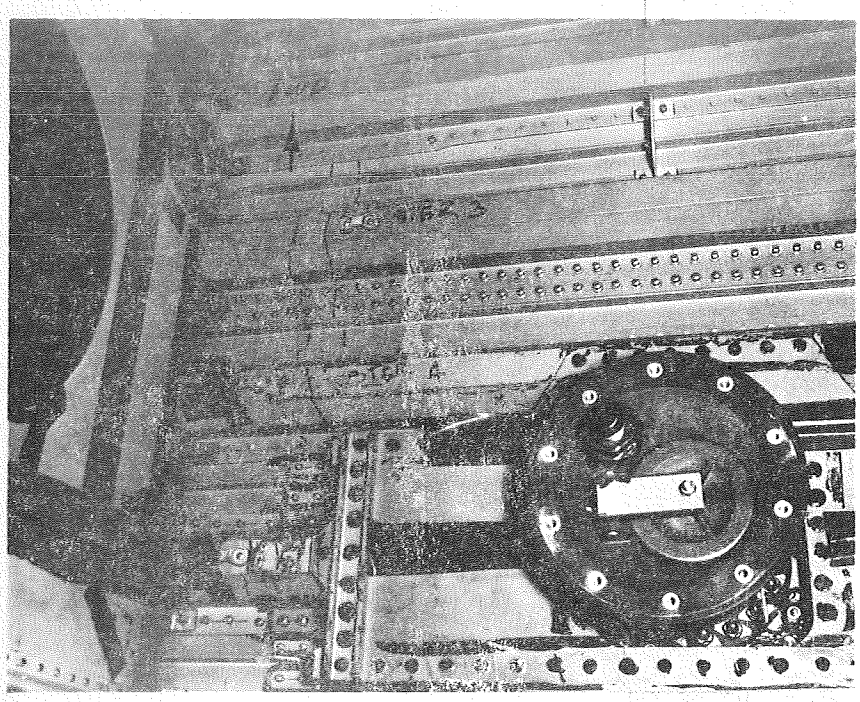


Figure 12-3 Stringers 3, 4 and 5 were broken during the crack propagation test, WS 105L.

RK 686-9

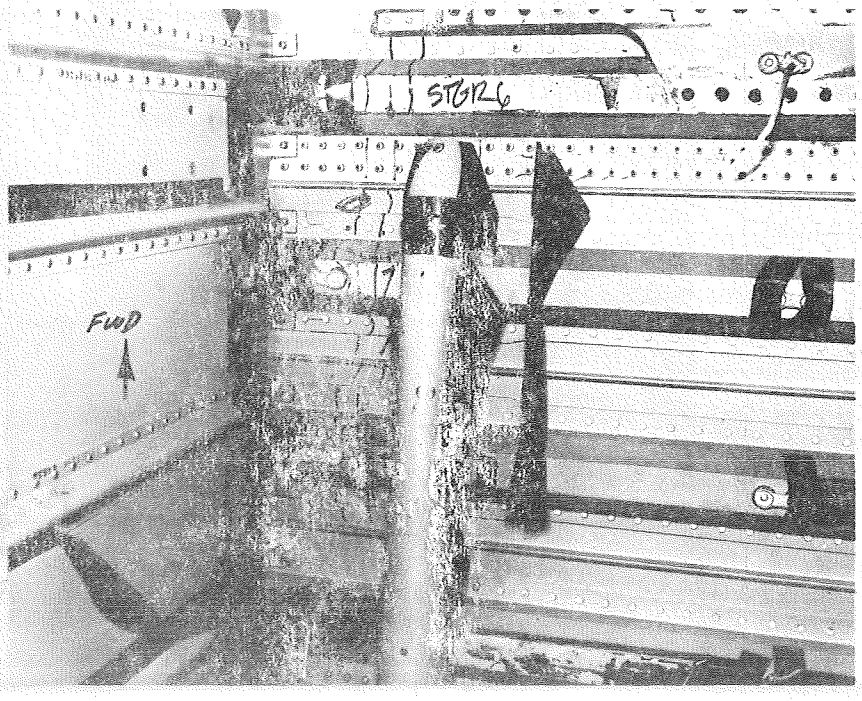
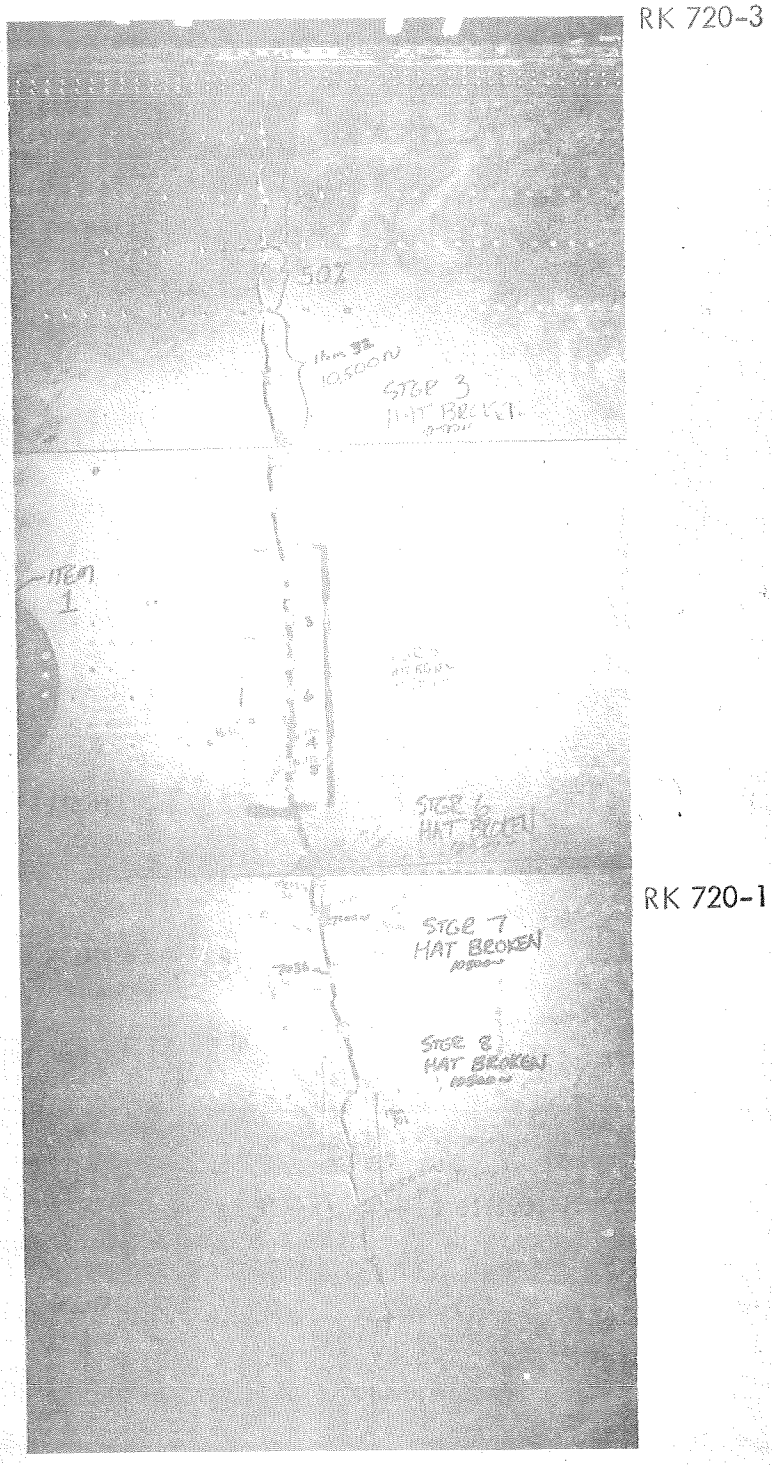


Figure 12-4 Stringers 6, 7 and 8 were broken during the crack propagation test, WS 105L.

RK 720-3



RK 720-1

Figure 12-5 Final failure occurred at 56% of limit load along WS 105L.

RK 0650-2

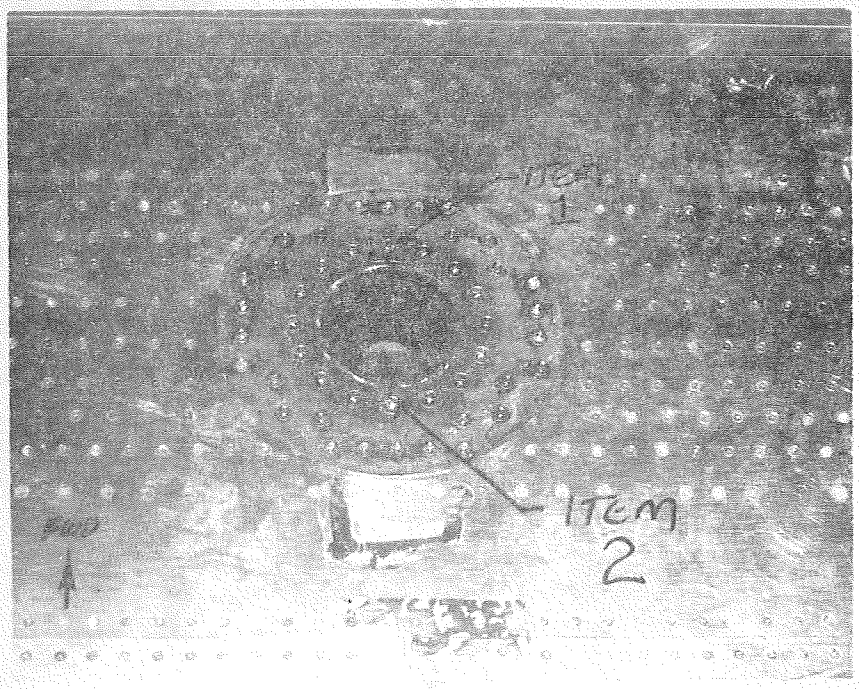


Figure 12-6 Skin cracks 1 and 2 originated in fastener holes common to the fuel filler fitting.

RK 686-2

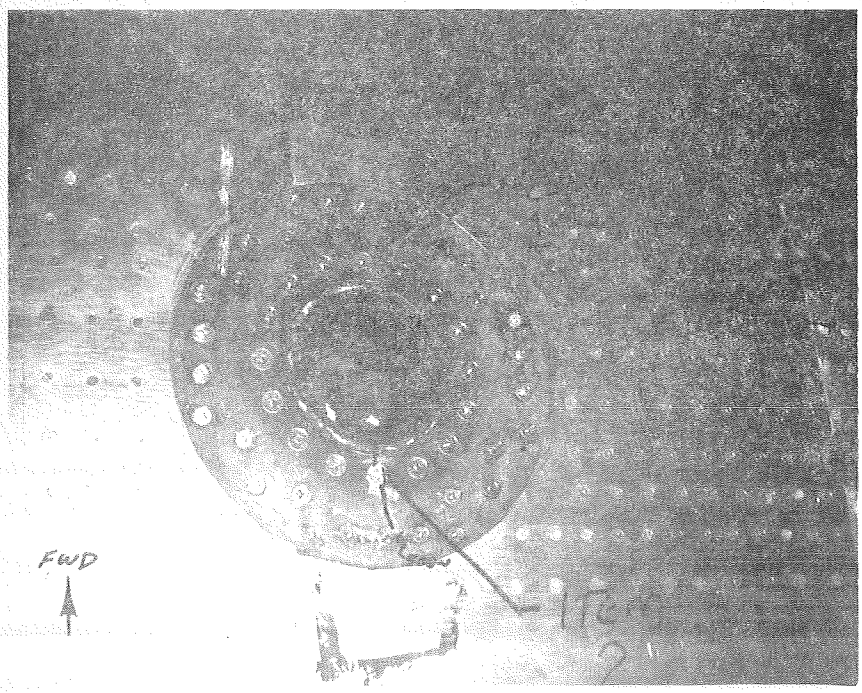


Figure 12-7 Crack extensions of cracks 1 and 2 as they appeared after 10,500 cycles.

RK 686-11

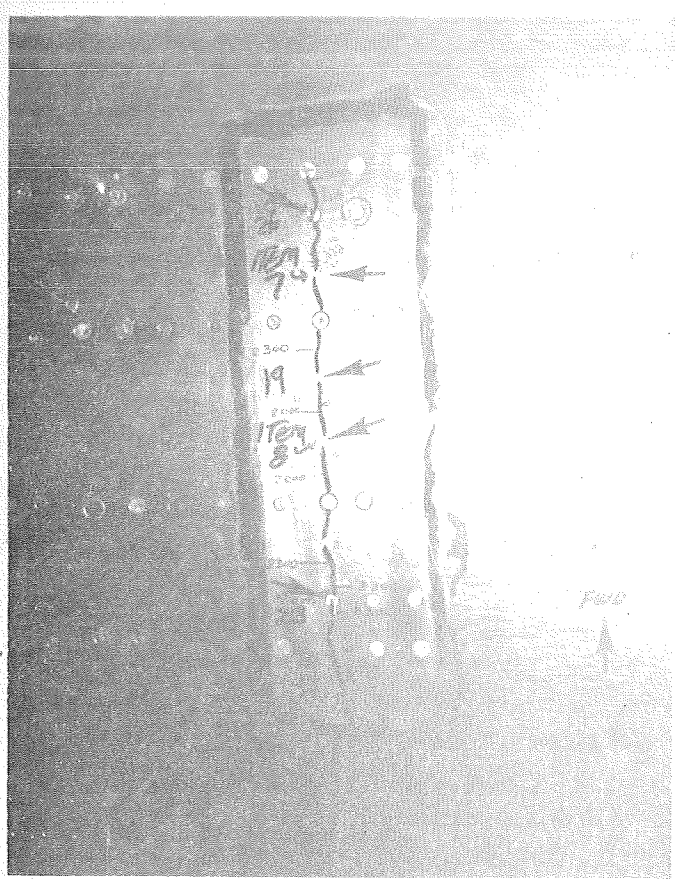


Figure 12-8

Cracked area, containing cracks 7, 8, 19, 23 and 26, is in the runout of the fuel filler cutout doubler at WS 135L (after 10,500 cycles).

RK 686-6

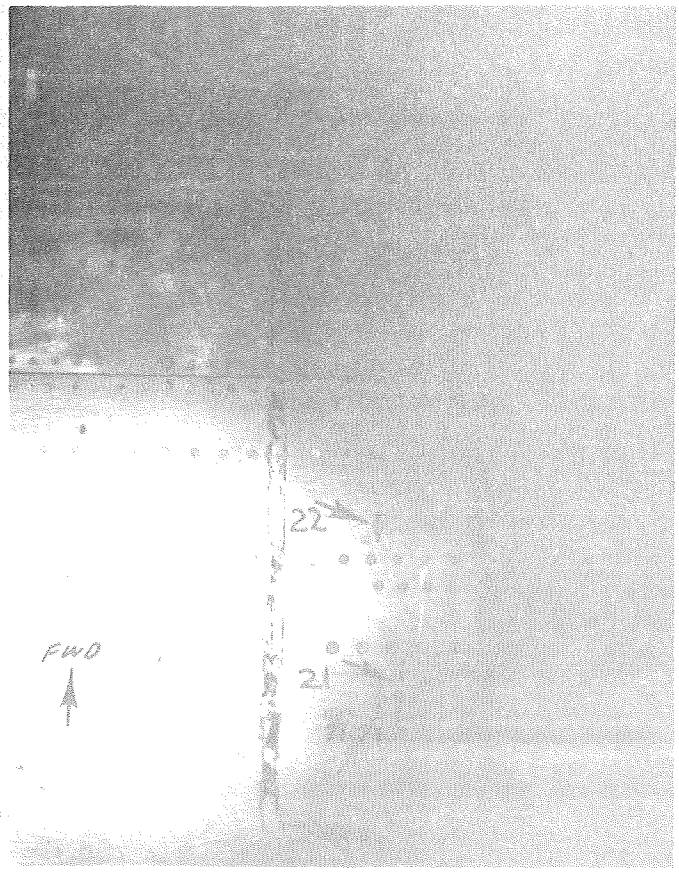


Figure 12-9

Cracks 21, 22 and 27 are located in the access door cutout doubler runout area at WS 36L.

RK 720-2

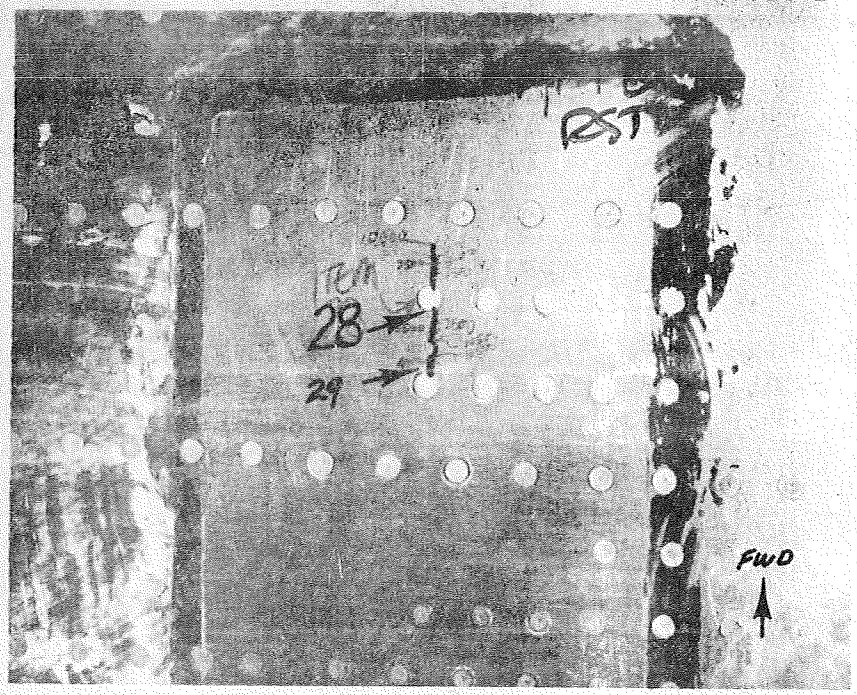


Figure 12-10 Cracks 28 and 29 are located in the access door cutout doubler runout area at WS 36L.

RK 686-10

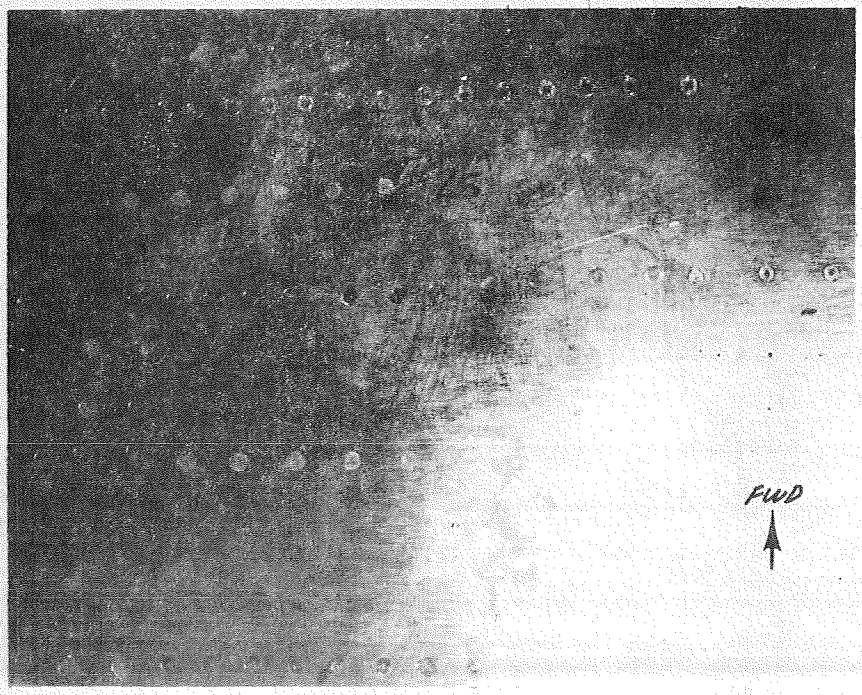


Figure 12-11 Cracks 24 and 25 are located at WS 174L near the aft inboard corner of the access door.

RK 0650-1

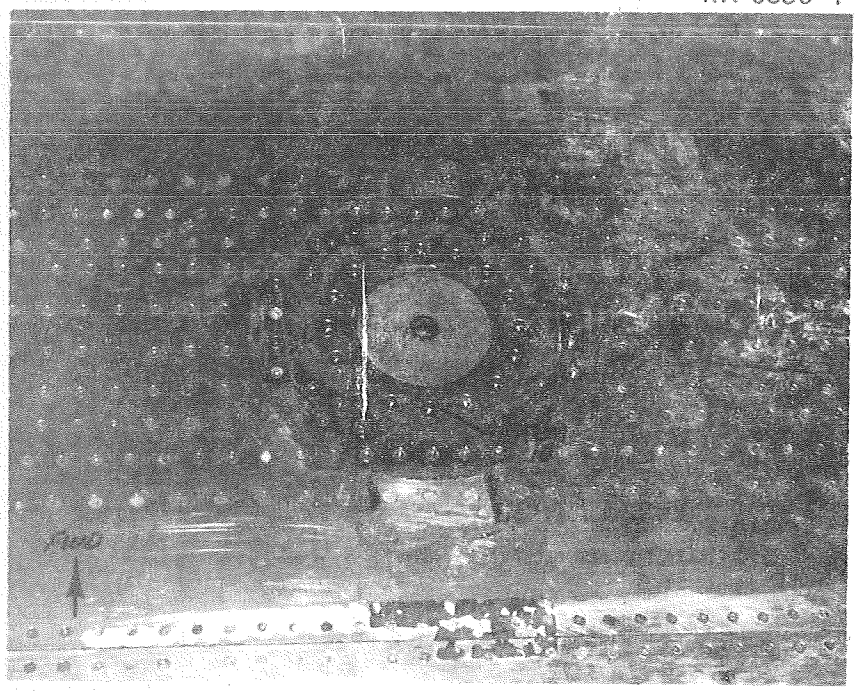


Figure 12-12 Skin cracks 9 and 10 originated in fastener holes common to the fuel filler fitting at WS 120.5R.

RK 686-5

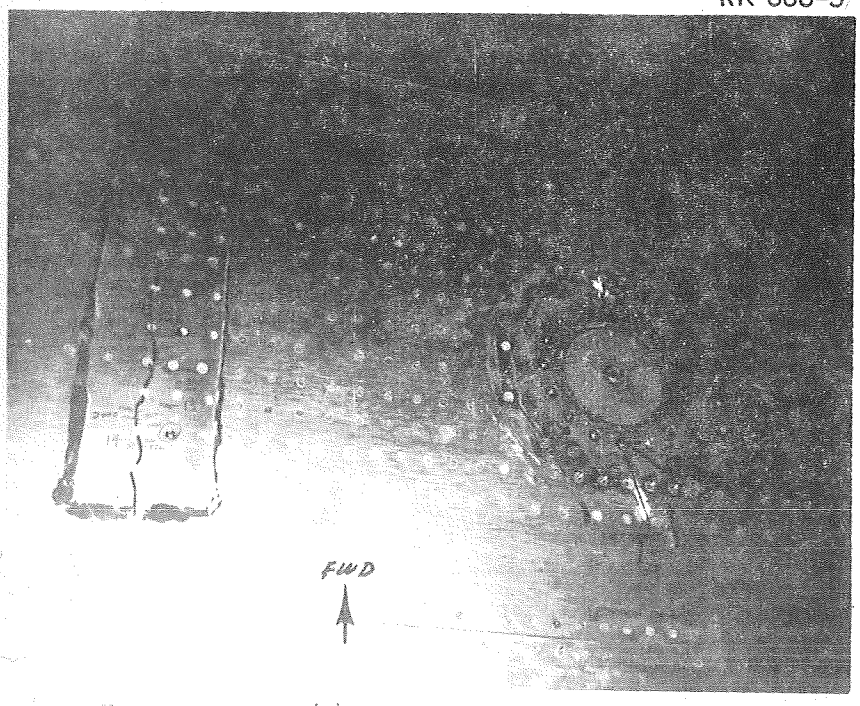


Figure 12-13 (1) Crack extension of 9 and 10 in the cutout area.
(2) Cracks 11, 12, 14, 15 and 20 in the doubler runout area at WS 105R.

RK 686-3

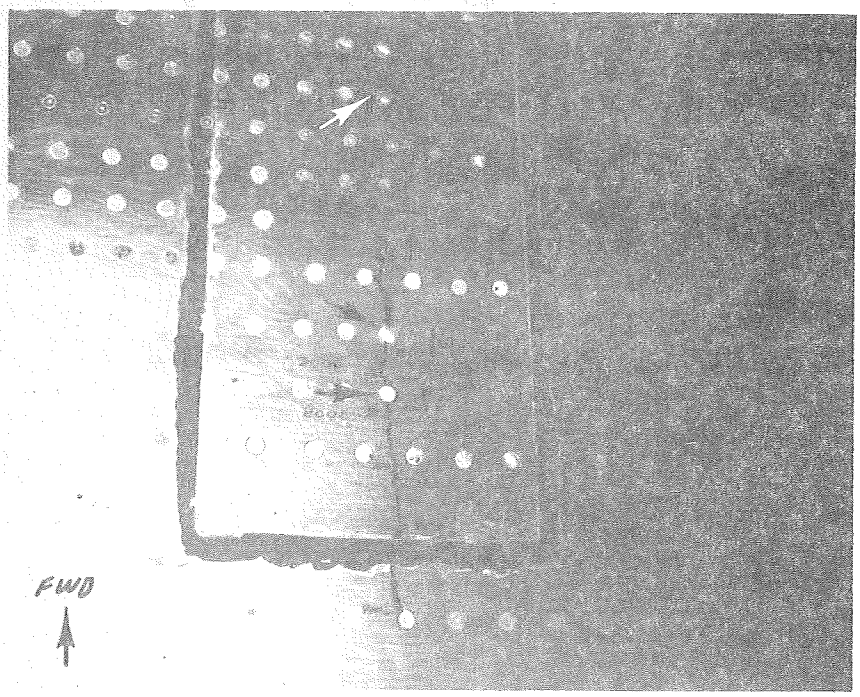


Figure 12-14 Skin cracks 16, 17 and 31 are located in the access door cutout doubler runout area at WS 35R.

PART 13

Specimen 13

CRACK PROPAGATION & RESIDUAL STRENGTH TEST RESULTS

TEST RESULTS

This was the first commercial C-130 airplane wing box tested, and had accumulated approximately 13,000 flight hours. X-ray examination confirmed field reports of a 9.4-inch skin crack in the center panel at WS 58R (Figure 13-22) so that patch was removed to simulate the unreinforced damaged condition during testing. This large crack (number 15) terminated at a fastener hole and at the edge of a panel. Another crack (number 16) of 3.3 inches existed at the same wing station at a different chordwise location (Figure 13-21); both ends of which terminated in holes.

The initial cyclic stress level was set at 35% Limit Load. After 10,000 cycles, nether the 3.3-inch or the 9.4-inch initial cracks had propagated. Cyclic load level was reduced slightly to 32-1/2% Limit Load. After 12,000 cycles (total), the 9.4-inch crack had grown to 10.8 inches and it continued to grow gradually to 16.8 inches at 20,000 cycles; the two underlying hat stringers were also cracked. The 3.3-inch crack grew to 3.8 inches at 18,000 cycles, and increased to 5.2 inches after 19,500 cycles (see Figures 13-23 and 13-24). Comprehensive crack history appears in Table 1-13 and Table 11-13 gives details on minor cracks.

During the static residual strength test detectable sonic reports and subsequent measurements of appreciable crack propagation occured at 42.7%, 44.4%, and 52.3% Limit Load. At this point approximately two-thirds of the central panel and one-third of the forward panel were cracked (out of a total of three lower surface panels), and at least two hat section stringers were cracked. At 62.3% Limit Load, a very large sonic report was heard, and was accompanied by a load system "dump" (load relief); a safety feature activitated by large deformation which normally occurs upon specimen failure. A subsequent inspection disclosed that all of the center panel, the forward panel, and the nine underlying hat stringers were severed at WS 58L. However, the aft panel and both the front and rear spars appeared to be intact. The dump was evidently caused by the sudden change in specimen deformation associated with panel failure, causing the control system safety feature to be activated. Since a significant portion of the structure appeared to be capable of supporting load, a decision was made to continue the test. At 20% Limit Load, there was a 1/4 inch to 1/2 inch surface separation along the main crack in Panels #2 and #3. At 53% Limit Load, the aft panel failed, and the automatic load relief device was again activated. Even with the entire surface structure - panels and reinforcing hat stringers - failed, the spars still appeared to remain intact; although a majority of the fasteners attaching the beam caps to the panel skins had failed in shear. Figures 13-25, 13-26 and 13-32 show the area of this failure. Accordingly, it was decided to terminate the test. Maximum residual strength after cyclic testing is recorded as the 62.3% Limit Load reached earlier at failure of the center and forward panels. General cracking in the outboard region of the left wing is shown in Figures 13-1, 13-2, 13-3, 13-4, 13-5, 13-6, 13-7, 13-8, 13-9, 13-10, and 13-11. Cracking around the corners of the WS 120.5 cutout on the left wing is shown in Figures 13-12, 13-13, 13-14, 13-15, and 13-16. Cracking at the corresponding right wing location is shown in Figures 13-17, 13-18, 13-19, and 13-20. General cracking in the outboard region of the right wing is shown in Figures 13-27, 13-28, 13-29, 13-30 and 13-31.

TABLE I-13

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 13

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 35% Limit Load — Number of Cycles (d)								
					0	2000	3000	4000	5000	6000	7000	8000	9000
40	The crack originated under an external "slingshot" doubler repair at the end of the internal reinforcing beam of the access door cutout	181.4L	2.0" aft of the stringer 15 ℄	13-7 13-11									
3	The crack originated under an external "slingshot" doubler repair at the end of the internal reinforcing beam of the access door cutout	181L	2.0" aft of the stringer 16 ℄	13-1 13-7 13-11	2.4	2.6	2.8	3.0	3.2	3.4	3.4	3.5	3.5
6	The crack originated in the edge of a drain hole in the skin	179L	2.9" aft of the stringer 19 ℄ a b	13-2 13-9 13-11	0.2 0.2(c)	—	—	—	—	0.2 0.2	0.5 0.1(c) 0.2	0.9 0.1 0.5	0.9 0.1 0.5
4	The crack originated in the edge of a fastener hole common to a stringer flange and the skin immediately adjacent to a repair	181L	1.8" fwd of the stringer 20 ℄ a b	13-2 13-9 13-11	1.0 0.3 0.4	1.2 0.4 0.4	1.2 0.4 0.4	1.4 0.6 0.4	1.5 0.7 0.4	1.6 0.8(c) 0.4(h)	— — —	— — —	1.6 0.8 0.4
5	Same as 3 except the crack emerged from both sides of the repair doubler	180.4L	1.0" fwd & aft of the stringer 20 ℄ a d	13-2 13-9 13-11	3.3 0.4 0.8	3.3 —	3.3 —	3.3 0.4 —	3.4 0.6 —	3.4 —	3.4 —	3.4 0.8	3.5 0.6 0.9
30	Same as 6	173.5L	2.9" aft of the stringer 13 ℄ a b	13-5 13-6 13-11									
7	Same as 6	173.5L	2.8" fwd of the stringer 18 ℄ a b	13-4 13-8 13-9 13-11	0.6 0.2 0.2	0.6 0.2 0.2	0.6 0.2 0.2	0.7 0.3 0.2	0.7 0.3 0.2	0.7 0.3 0.2	0.8 0.3 0.3	0.8 0.3 0.3	0.8 0.3 0.3
20	The crack originated in the edge of fastener holes common to the internal reinforcing beam for the access door cutout and the skin near a repair	179.5R	0.5" fwd & aft of the stringer 21 ℄ a d	13-30	3.4 1.0 1.0(c)	—	—	—	—	—	—	—	3.4 1.0 1.0
39	The crack originated under the engine thrust reaction angle and runs under an external repair doubler	180R	Visible portion in the stringer 18 area	13-31									

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-13 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 13

LOWER SURFACE SKIN, (a)

Crack No.	Upbending, Phase 2 Cyclic Test, R = 0.1 Max. Load = 32.5% Limit Load — Cumulative Number of Cycles (d)										Upbending Residual Strength Test				
	11,000	12,000	13,000	14,000	15,000	18,000	18,500	19,000	19,500	20,000	42.7%	44.4%	52.3%	62.3%	53%
40										0.6				0.6	1.1
3	3.7	3.8	3.8	4.1	4.2	4.6	4.7	4.8	4.8	4.9				4.9	5.2
6	0.9				0.9	1.3	1.3	1.3	1.4						1.4
a	0.1				0.1	0.5	0.5	0.5	0.6						0.6
b	0.5				0.5	0.6	0.6	0.6							0.6
4	1.6														1.6
a	0.8														0.8
b	0.4														0.4
5	3.8	3.9													3.9
a	0.6	0.7													0.7
b	1.1	1.2(b)													1.2
30 a					0.7	0.8	0.8	1.0						1.0	1.0
b					0.2	0.3	0.3	0.4						0.4	0.4
					0.2	0.2	0.2	0.4						0.4	0.4
7	1.1	1.1	1.1	1.1	1.3	1.5	1.7	1.7	1.7	1.7				1.7	1.8
a	0.4	0.4	0.4	0.4	0.5	0.6	0.7	0.7	0.7	0.7				0.7	0.8
b	0.4	0.4	0.4	0.4	0.5	0.6	0.7	0.7	0.7	0.8				0.8	0.8
20	3.4				3.4	3.5									3.5
a	1.0				1.0	1.2									1.2(c)
d	1.0				1.0	1.0									1.0
39											4.5				4.5

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-13 (Cont.)
SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 13
LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 35% Limit Load — Number of Cycles (d) 0	2000	3000	4000	5000	6000	7000	8000	9000
8	The crack originated in the edge of the last fastener hole common to the internal reinforcing doubler of the access door cutout & the skin	168L	The stringer 17 ℄	13-3	0.2					0.2	0.3	0.3	0.3
			a	13-8	0.2					0.2	0.3	0.3	0.3
			b	13-11									
9	Same as crack 8	168L	The stringer 18 ℄	13-4	0.2					0.2	0.3	0.3	0.3
			a	13-8	0.2					0.2	0.3	0.3	0.3
			b	13-11									
24	Same as crack 8, except the fastener hole was also common to the stringer flange	169R	1.8" aft of the stringer 18 ℄	13-29								0.7	0.7
			a									0.3	0.3
			b									0.1	0.1
11	The crack originated at a door attachment fastener	128L	0.8" fwd of the stringer 17 ℄	13-12	1.6								1.6
			a		0.6(c)								0.6
			b		0.3(b)								0.3
12	Same as crack 11	128L	0.8" aft of the stringer 19 ℄	13-13	1.6								1.6
			a		0.3(b)								0.3
			b		0.6(c)								0.6
13	Same as crack 11	113L	0.8" fwd of the stringer 17 ℄	13-14	0.1	0.1	0.1	0.1	0.3	0.3	0.3	0.3	1.1
			a	13-16									0.1
			b		0.1	0.1	0.1	0.1	0.3(b)	0.3	0.3	0.3	0.3
14	Same as crack 11	113L	0.8" aft of the stringer 19 ℄	13-15	1.6								1.6
			a		0.3(b)								0.3
			b		0.6(c)								0.6
27	Same as crack 11	110.5L	0.3" fwd of the stringer 17 ℄	13-16									
31	Same as crack 11	112L	0.8" fwd of the stringer 17 ℄	13-16									
17	Same as crack 11	113R	0.8" fwd of the stringer 17 ℄	13-17	1.2	1.2	1.2	1.3	1.3	1.3	1.4	1.5	1.5
			a		0.2	0.2	0.2	0.3	0.3	0.3	0.5	0.5	0.5
			b		0.3(b)	0.3	0.3	0.3	0.3	0.3	0.3	0.3	0.3

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-13 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 13

LOWER SURFACE SKIN (a)

Crack No.		Upbending, Phase 2 Cyclic Test, R = 0.1 Max. Load = 32.5% Limit Load — Cumulative Number of Cycles (d)										Upbending Residual Strength Test				
		11,000	12,000	13,000	14,000	15,000	18,000	18,500	19,000	19,500	20,000	42.7%	44.4%	52.3%	62.3%	53%
8			0.3	0.4	0.4	0.9	1.0	1.0	1.0	1.2						1.2
	a		0.3	0.4	0.4	0.5	0.5	0.5	0.5	0.6						0.6
	b					0.1	0.1	0.1	0.1	0.2						0.2
9		0.4			0.4	0.9	0.9			0.9	1.1					1.1
	a	0.4			0.4	0.4	0.5			0.5	0.6					0.6
	b					0.1	0.1			0.1	0.2					0.2
24		0.7	0.8	0.8	0.8	1.0	1.0	1.2	1.2	1.2	1.3					1.3
	a	0.3	0.3	0.3	0.3	0.4	0.4	0.5	0.5	0.5	0.5					0.5
	b	0.1	0.2	0.2	0.2	0.3	0.3	0.4	0.4	0.4	0.5					0.5
11		1.6														1.6
	a	0.6														0.6
	b	0.3														0.3
12		1.6														1.6
	a	0.3														0.3
	b	0.6														0.6
13		1.2	1.2	1.2	1.4	1.4	1.6									1.6
	a	0.2	0.2	0.2	0.4	0.4	0.6(c)									0.6
	b	0.3	0.3	0.3	0.3	0.3	0.3									0.3
14		1.6														1.6
	a	0.3														0.3
	b	0.6														0.6
27					0.2	0.9										0.9
	a					0.1										0.1
	b				0.2	0.2										0.2
31						1.0				1.0	1.2					1.2
	a					0.3				0.3	0.4					0.4
	b					0.2				0.2	0.2					0.2
17		1.5	1.6													1.6
	a	0.5	0.7(c)													0.7
	b	0.3	0.3													0.3

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-13 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 13

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 35% Limit Load — Number of Cycles (d) 0	2000	3000	4000	5000	6000	7000	8000	9000
18	Same as crack 11	113R	0.8" aft of the stringer 19 ℄	13-18	0.3				0.3	1.0	1.0	1.0	1.1
			a		0.3(b)				0.3	0.3	0.3	0.3	0.3
			b						0.1	0.1	0.1	0.1	0.2
19	Same as crack 11	128R	0.8" aft of the stringer 19 ℄	13-18	0.3	0.3	0.3	0.3	1.0	1.0	1.1	1.1	1.1
			a	13-20	0.3(b)	0.3	0.3	0.3	0.3	0.3	0.3	0.3	0.3
			b						0.1	0.1	0.1	0.1	0.1
15	The crack originated in the last fastener holes inboard common to the reinforcement beam for the access door cutout and the skin	58L	0.5" fwd and aft of the stringer 15 ℄	13-21	3.3								3.3
			a	13-23	1.0(c)								1.0
			d	13-25	0.9(c)								0.9
16	Same as crack 15	58L	0.5" fwd and aft of the stringer 16 ℄	13-22	9.4								9.4
			a	13-23	1.6(b)								1.6
			d	13-25	6.5(c)								6.5
				13-26									

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-13 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 13

LOWER SURFACE SKIN, (a)

Crack No.	Upbending, Phase 2 Cyclic Test, R = 0.1 Max. Load = 32.5% Limit Load — Cumulative Number of Cycles (d)										Upbending Residual Strength Test				
	11,000	12,000	13,000	14,000	15,000	18,000	18,500	19,000	19,500	20,000	42.7%	44.4%	52.3%	62.3%	53%
18	1.1			1.1	1.3	1.4	1.4	1.4	1.5					1.5	1.6
a	0.3			0.3	0.3	0.3	0.3	0.3	0.3					0.3	
b	0.2			0.2	0.4	0.5	0.5	0.5	0.6					0.6	0.6(c)
19	1.1	1.2	1.2	1.2	1.4	1.5								1.5	1.6
a	0.3	0.3	0.3	0.3	0.3	0.3								0.3	0.3
b	0.1	0.3	0.3	0.3	0.5	0.6								0.6	0.6(c)
15	3.3				3.3	3.8	4.0	4.6	5.2	5.2	7.6	9.1	9.1	25.9	↑
a	1.0				1.0	1.0	1.0	1.0	1.0	1.0	4.5	6.0(c)	6.0	22.7(b)	
b	0.9				0.9	1.5	1.7	2.4	2.9(b)	2.9	2.9			2.9	Complete Chordwise Failure
16	9.4	10.8	11.9	12.6	13.5	13.8	14.0	14.6(k)	15.2	16.8	17.7	17.7	20.9	27.2	
a	1.6													1.6	
d	6.5	7.9	9.0	9.7	10.6(c)	10.9	11.1	11.7	12.3	13.9	14.8	14.8	18.0	24.3	↓

NOTE: Footnotes found in Table III of the General Section, Part G.

GA FORM 425A-1

TABLE II-13

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 3

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta., In.	Crack Location	Fig. No.	Cycles When Discovered	Length When Discovered, In.	(d) Length After 20,000 Cycles	Length After Failure at 53% Limit Load
1	The crack originated in the steel engine exhaust heat shield in the edge of a fastener hole common to the heat shield and the skin	187L	2.7" aft of the stringer 21 ℄	13-9	0	0.6	0.6	0.6
			a			0.3(b)	0.3	0.3
			b			0.1	0.1	0.1
2	The crack originated in Panel No. 2, in the edge of a fastener hole common to the panel splice (flange of stringer 16) and the skin	181L	1.8" fwd of the stringer 16 ℄	13-1	0	0.3	0.3	0.3
			a	13-7		0.3(b)	0.3	0.3
			b	13-11		-	-	-
10	The crack originated in the edge of a fastener hole common to the access door cutout doubler, a stringer flange and the skin	168L	1.8" aft of the stringer 18 ℄	13-3	0	0.1	0.1	0.1
			a	13-2		0.1	0.1	0.1
			b	13-11		-	-	-
21	Same as crack 1	187R	2.7 aft of the stringer 21 ℄	13-36	2,000	0.1	0.1	0.1
			a			0.1	0.1	0.1
						-	-	-
22	The crack originated in the edge of a drain hole in the skin	173.5R	2.7" fwd of the stringer 17 ℄	13-28	5,500	0.1	0.2	0.2
			a	13-31		0.1	0.2	0.2
			b			-	-	-
23	Same as crack 10	168L	1.8" fwd of the stringer 18 ℄	13-8	7,000	0.1	0.3	0.3
			a	13-11		0.1	0.3	0.3
			b			-	-	-
25	Same as crack 22	173.5L	2.7" fwd of the stringer 17 ℄	13-7	7,000	0.1	0.6	0.6
			a	13-11		0.1	0.1	0.1
			b			-	0.2	0.2
26	Same as crack 22	175L	2.9" aft of the stringer 20 ℄	13-9	11,000	0.5	0.6	0.6
			a	13-11		0.1	0.2	-.2
			b			0.2	0.2	0.2
28	Same as crack 22	173.5L	2.8" fwd of the stringer 15 ℄	13-6	14,500	0,7	0.9	0.9
			a	13-11		0.3	0.4	0.4
			b			0.1	0.2	0.2
29	The crack originated in a fastener hole common to the access door cutout reinforcement beam and the skin	177.5L	0.5" fwd of the stringer 16 ℄	13-7	14,500	0.1	0.1	0.1
			a	13-11		-	-	-
			b			0.1	0.1	0.1

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-13

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 13

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta., In.	Crack Location	Fig. No.	Cycles When Discovered	Length When Discovered, In.	(d) Length After 20,000 Cycles	Length After Failure at 58% Limit Load
32	Same as crack 22	175R	2.8" fwd of the stringer 12 ℄	13-27	14,500	0.7	0.9	0.9
			a	13-31		0.4	0.5	0.5
			b			0.1	0.2	0.2
33	The crack originated in the edge of a fastener hole common to the access door cutout doubler and the skin	168L	At the stringer 19 ℄	13-8	16,000	0.2	0.2	0.2
			a	13-11		0.2	0.2	0.2
			b			-	-	-
34	Same as crack 22	175.5L	2.8" fwd of the stringer 12 ℄	13-5	17,500	0.3	0.3	0.3
			a	13-11		0.3	0.3	0.3
			b			-	-	-
35	Same as crack 22	175.5L	2.8" aft of the stringer 23 ℄	13-10	17,500	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
36	Same as crack 22	173.5R	2.8" fwd of the stringer 13 ℄	13-27	17,500	0.1	0.1	0.1
			a	13-31		0.1	0.1	0.1
			b			-	-	-
37	Same as crack 22	173.5R	2.8" aft of the stringer 18 ℄	13-29		0.5	0.7	0.7
			a	13-31		0.2	0.3	0.3
			b			0.1	0.2	0.2
38	Same as crack 22	175R	2.8" fwd of the stringer 14 ℄	13-29	18,000	0.2	0.2	0.2
			a	13-31		0.2	0.2	0.2
						-	-	-

NOTE: Footnotes found in Table III of the General Section, Part G.

RK 0735-5

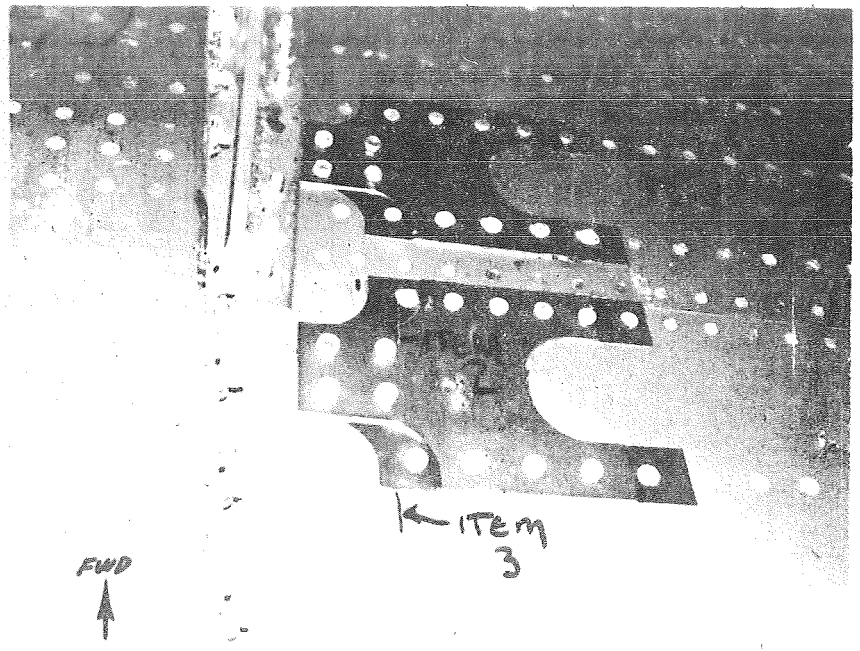


Figure 13-1 WS 176L, stringers 15 & 16 area. Prior to testing showing items 2 & 3.

RK 0735-2

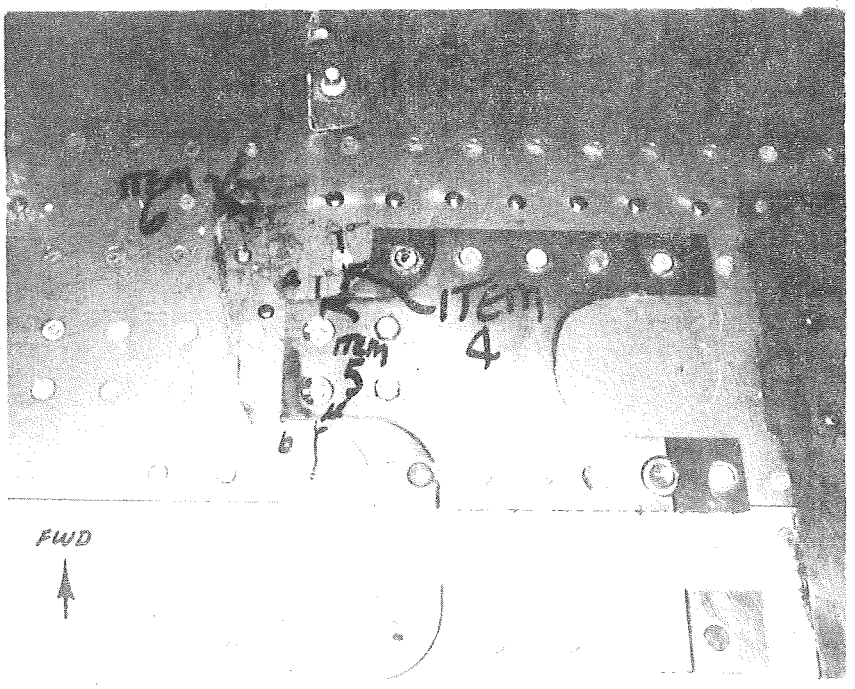


Figure 13-2 WS 176L, stringer 20 area, prior to testing, showing items 4, 5 & 6.

RK 0735-4

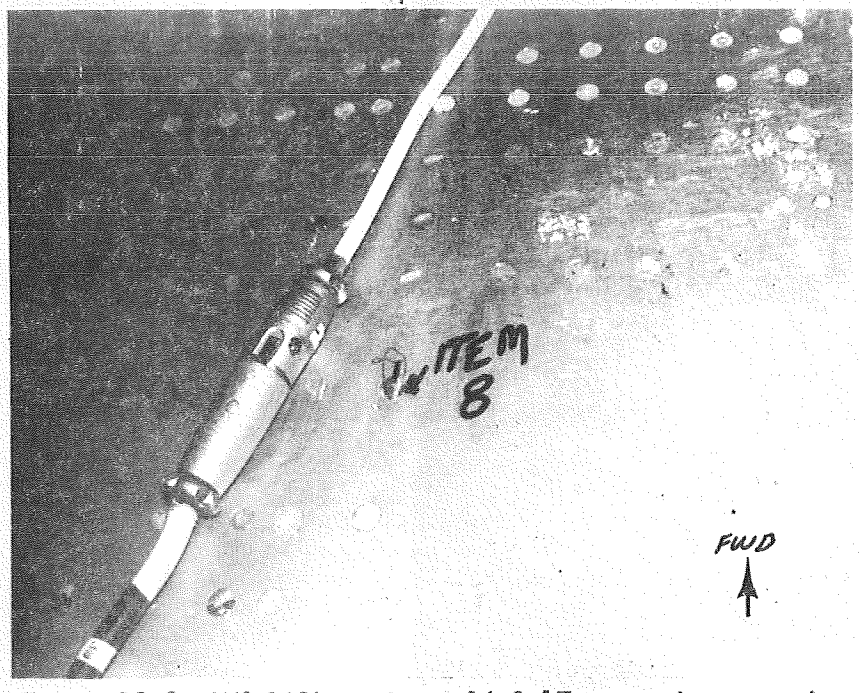


Figure 13-3 WS 168L, stringer 16 & 17 area prior to testing, showing item 8.

RK 0735-3

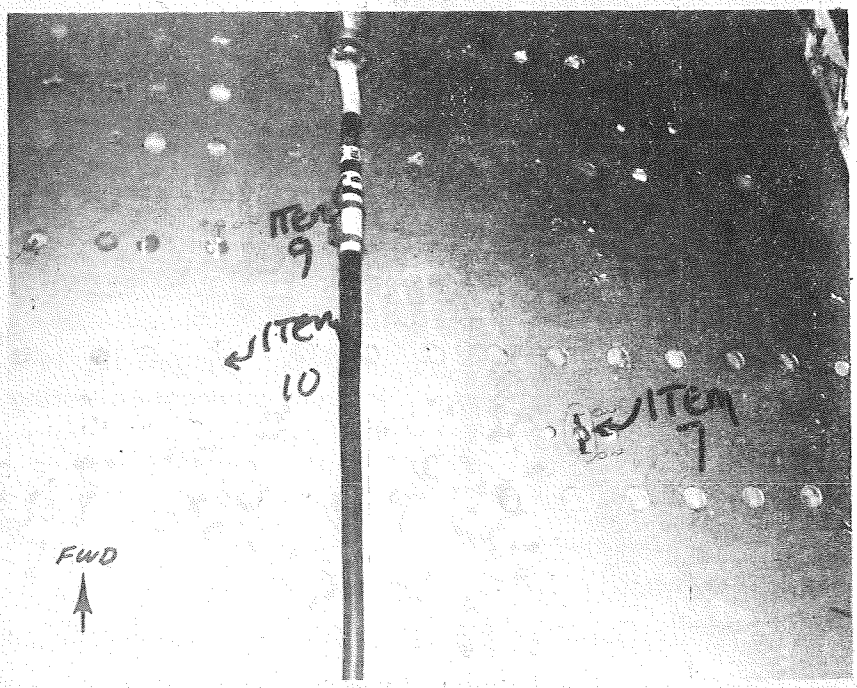
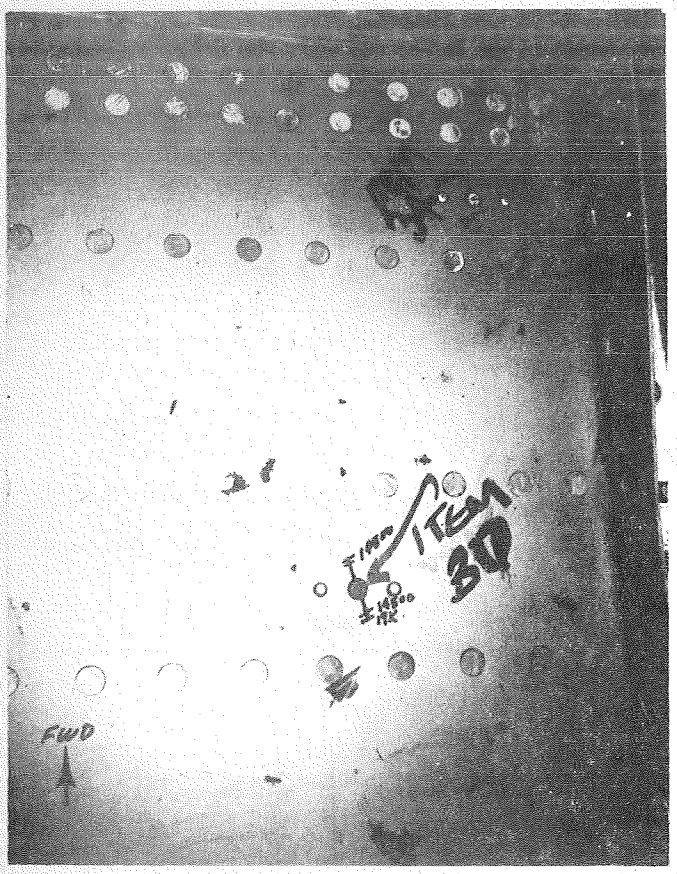


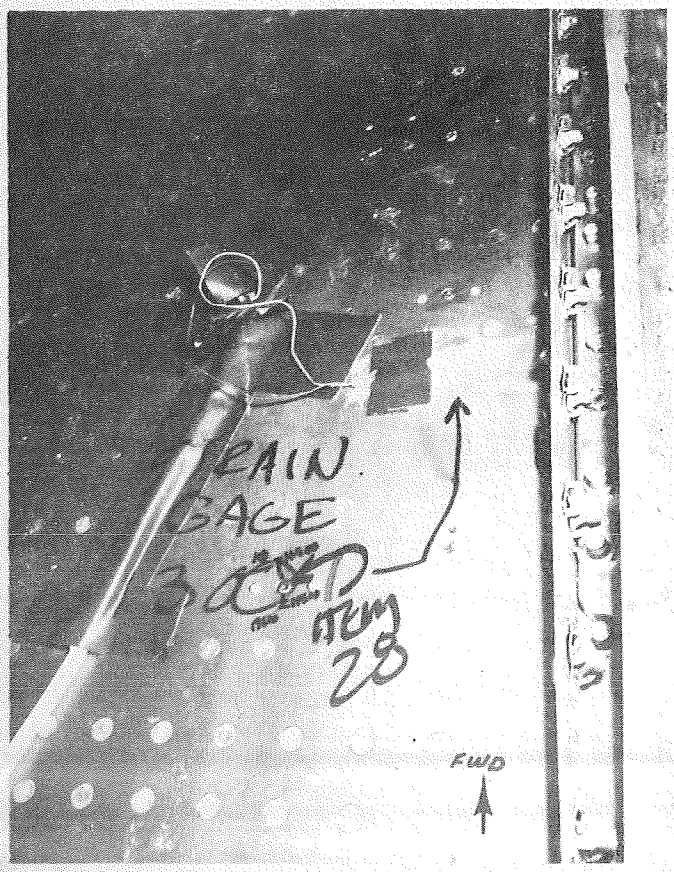
Figure 13-4 WS 168L, stringer 18 & 19 area prior to testing, showing items 9, 10 & 7.



RK 0898-17

Figure 13-5

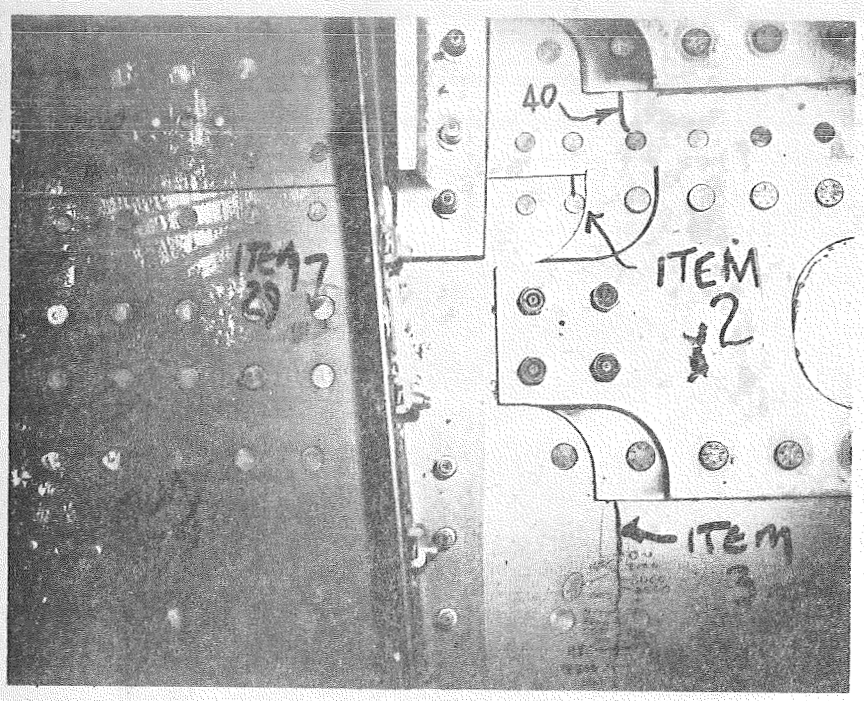
WS 170L, stringers 12, 13 area, subsequent to crack propagation testing showing items 34, 30.



RK 0898-16

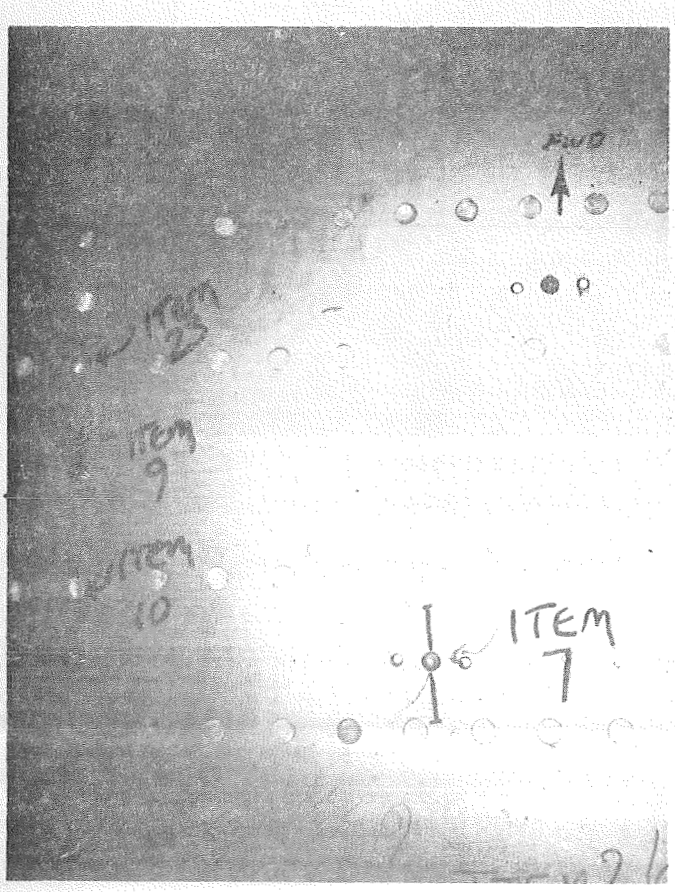
Figure 13-6

WS 170L, stringers 13, 14 & 15 area subsequent to crack propagation testing showing items 30, 28.



RK 0898-15

Figure 13-7 WS 170-176L stringer 16 area subsequent to crack propagation testing, showing items 2, 3, 29 and 25.



RK 0898-14

Figure 13-8

WS 168L stringer 17, 18 & 19 area subsequent to crack propagation testing, showing items 7, 8, 23, 9, 10, & 33.

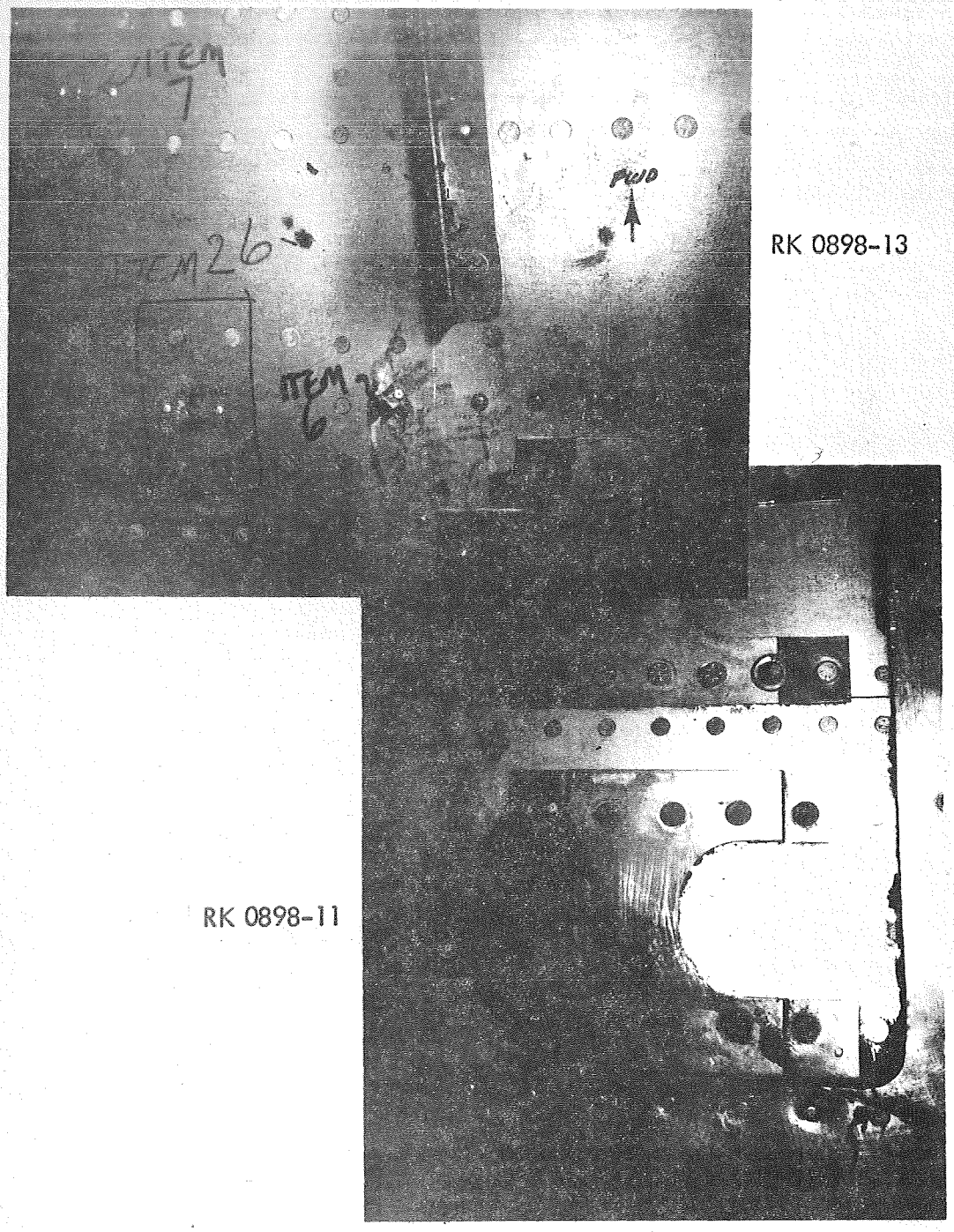


Figure 13-9 WS 176L, stringer 19, 20 & 21 area subsequent to crack propagation testing showing items 7, 26, 6, 4, 5, & 1.

RK 0898-12

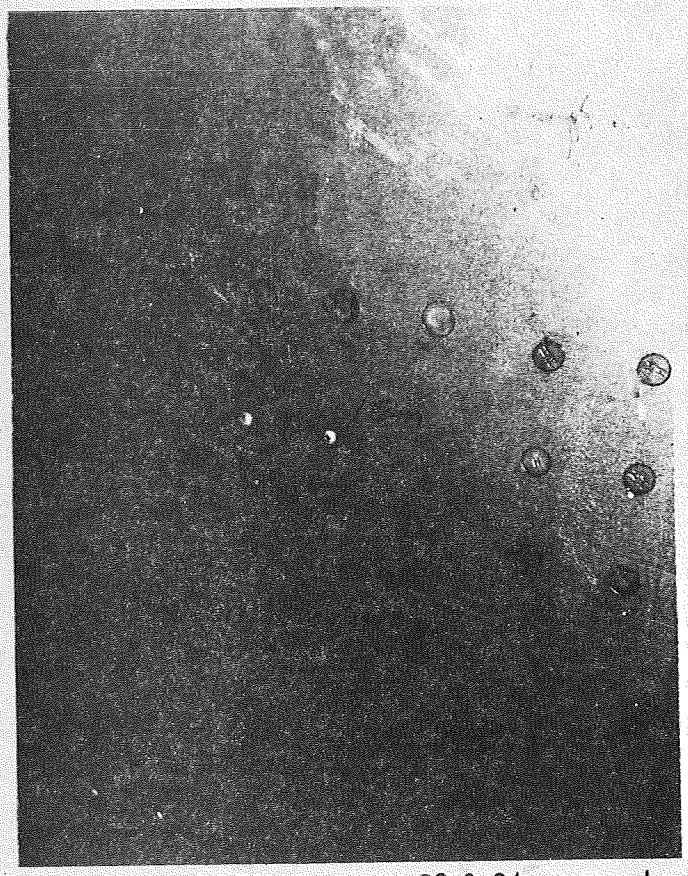


Figure 13-10 WS 175L, stringer 23 & 24 area, subsequent to crack propagation testing, showing item 35.

RK 0884-8

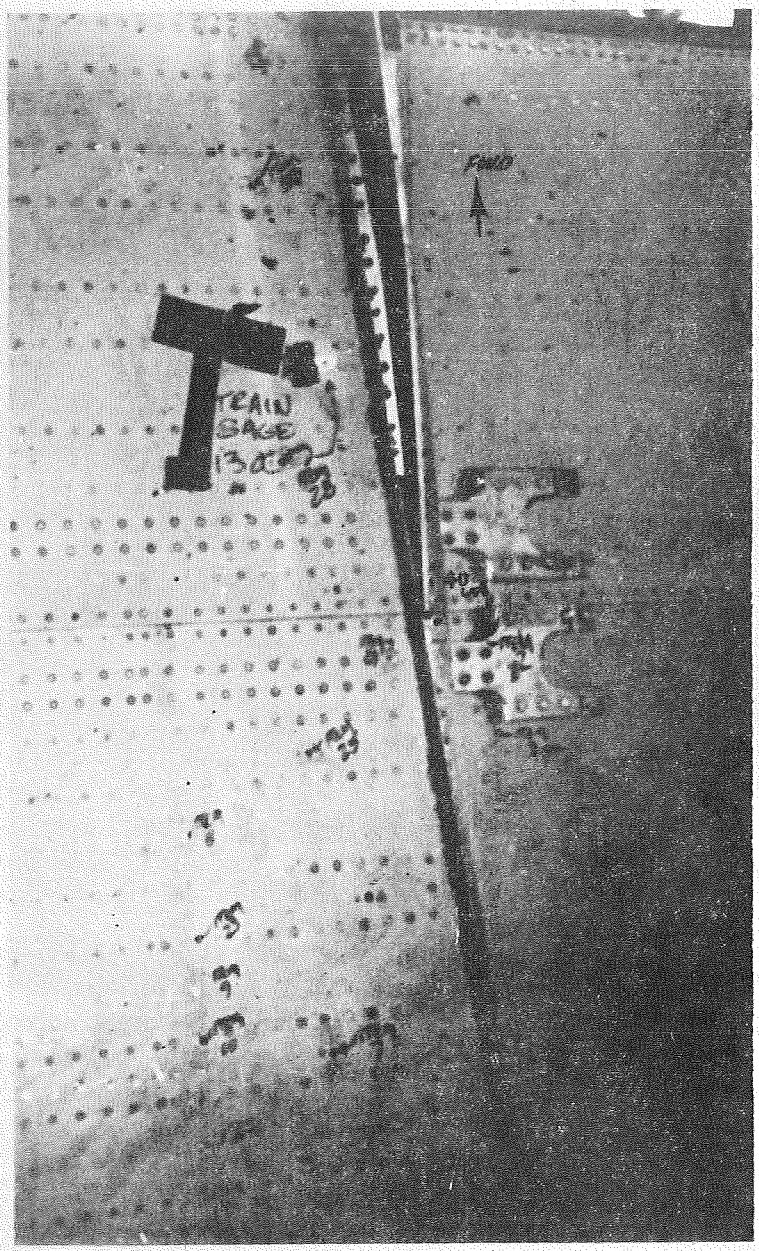


Figure 13-11 WS 176L - 168L area after final failure showing items 34, 30, 28, 29, 2, 3, 25, 8, 23, 9, 10, 33, 7, 26, 6, 4 & 5.

RK 0898-21

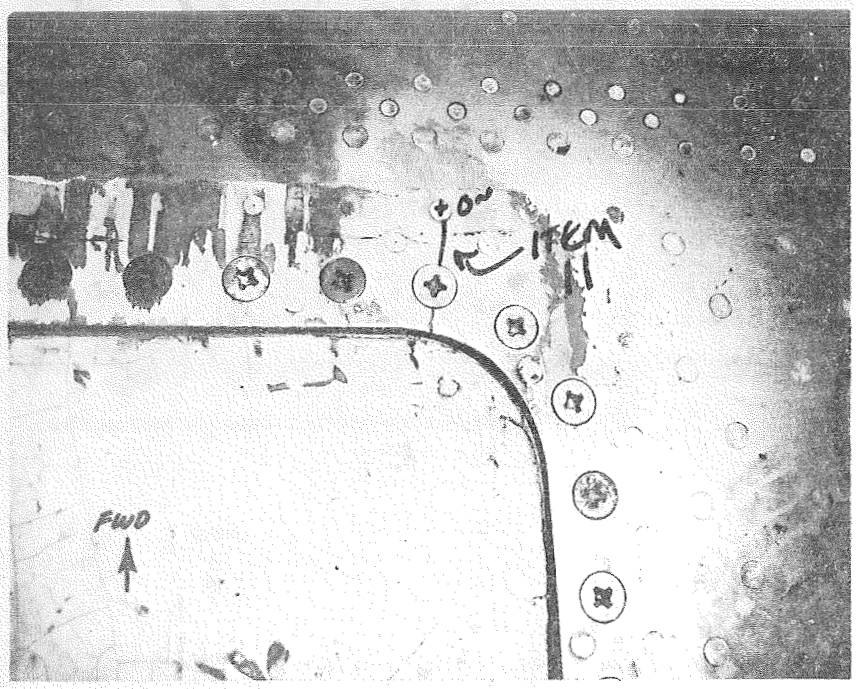


Figure 13-12 WS 128L, stringer 17 area. Fwd outboard corner of lower surface fuel bag access door, prior to testing, showing item 11.

RK 0898-20

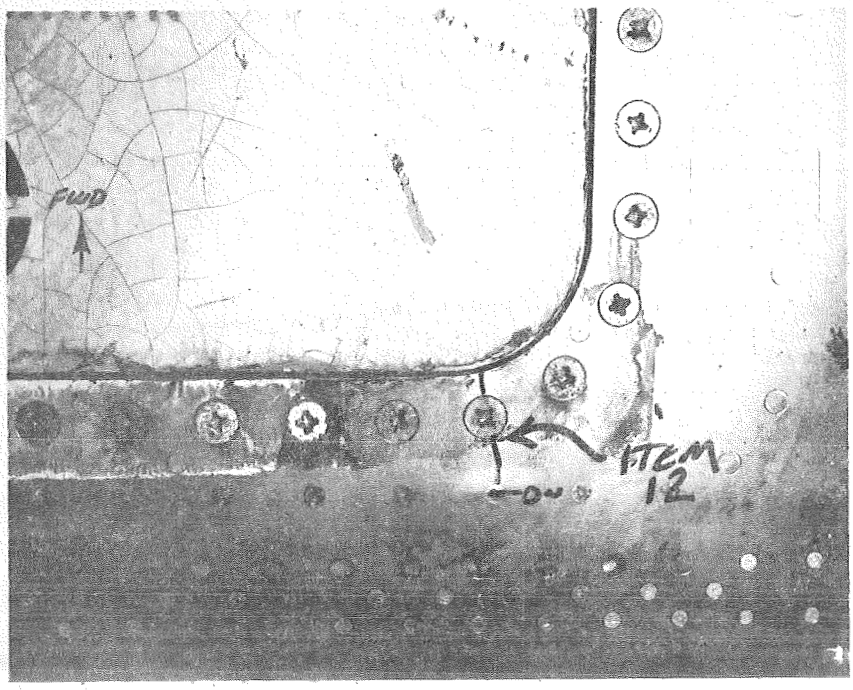


Figure 13-13 WS 128L, stringer 19 area, aft outboard corner of lower surface fuel bag access door, prior to testing, showing item 12.

RK 0735-9

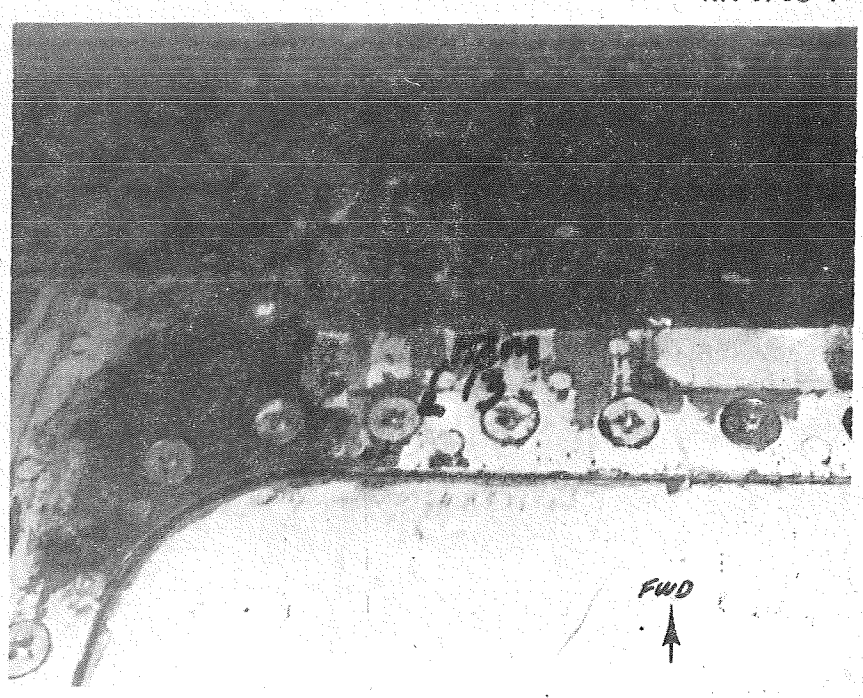


Figure 13-14 WS 112.8L, stringer 17 area, fwd inboard corner of lower surface fuel bag access door showing item 13.

RK 0898-18

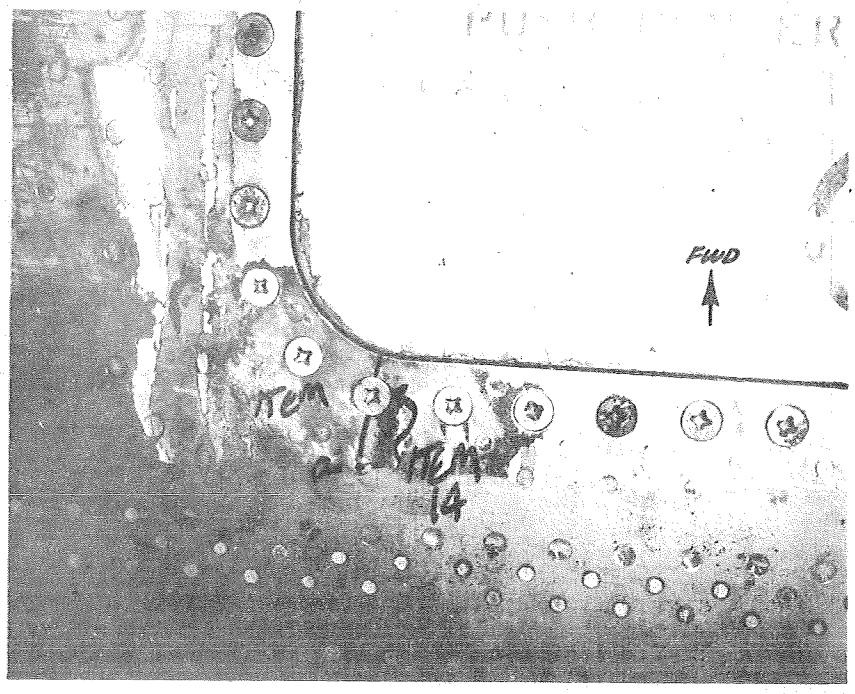


Figure 13-15 WS 112.8L stringer 19 area, aft inboard corner of lower surface fuel bag access door, showing item 14.

RK 0898-19

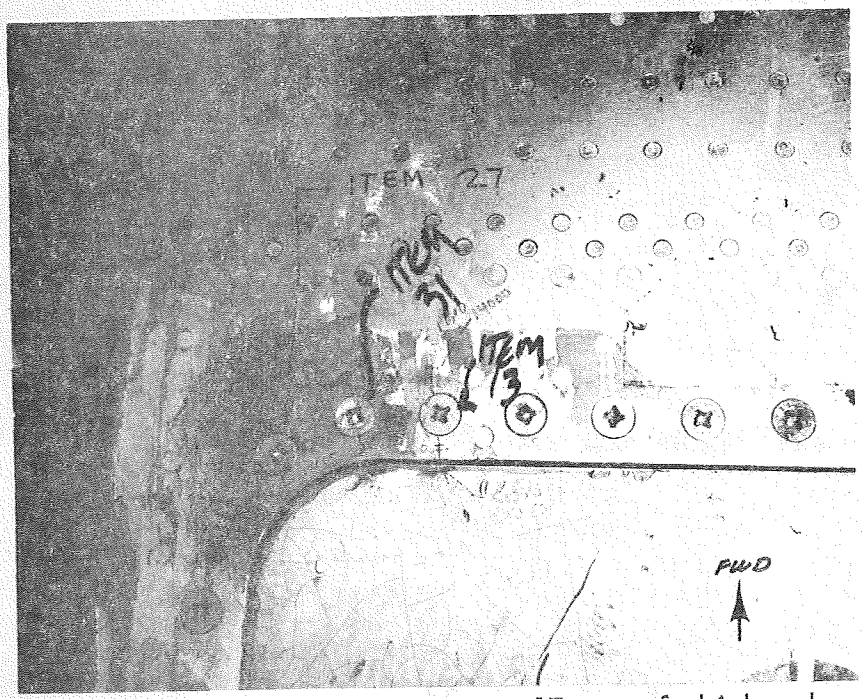


Figure 13-16 WS 112.8L, stringer 17 area, fwd inboard corner of lower surface fuel bag access door, subsequent to crack propagation testing, showing items 27, 31, & 13.

RK 0735-14

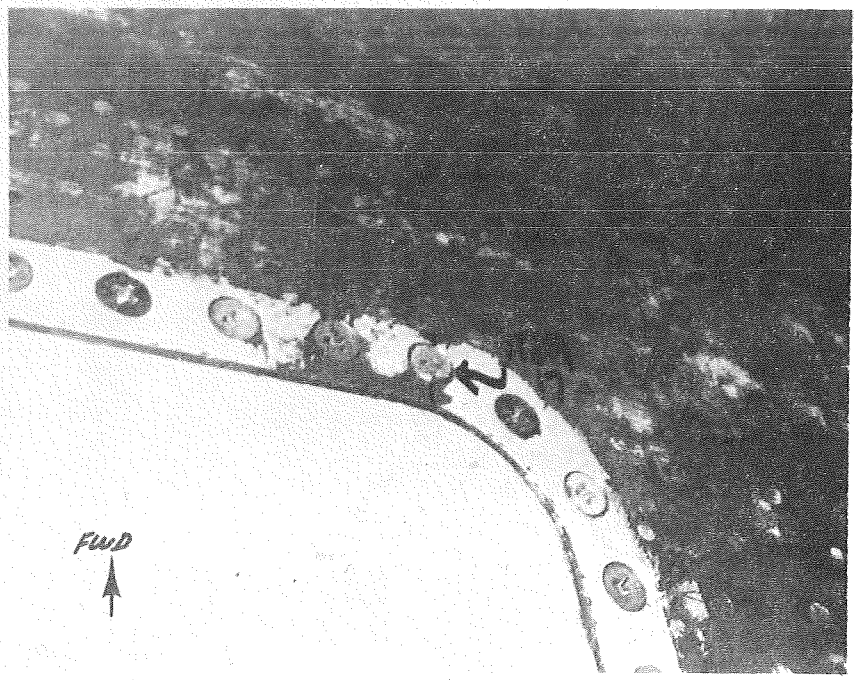


Figure 13-17 WS 112.8R, stringer 17 area, fwd inboard corner of lower surface fuel bag access door, prior to testing, showing item 17.

RK 0735-13

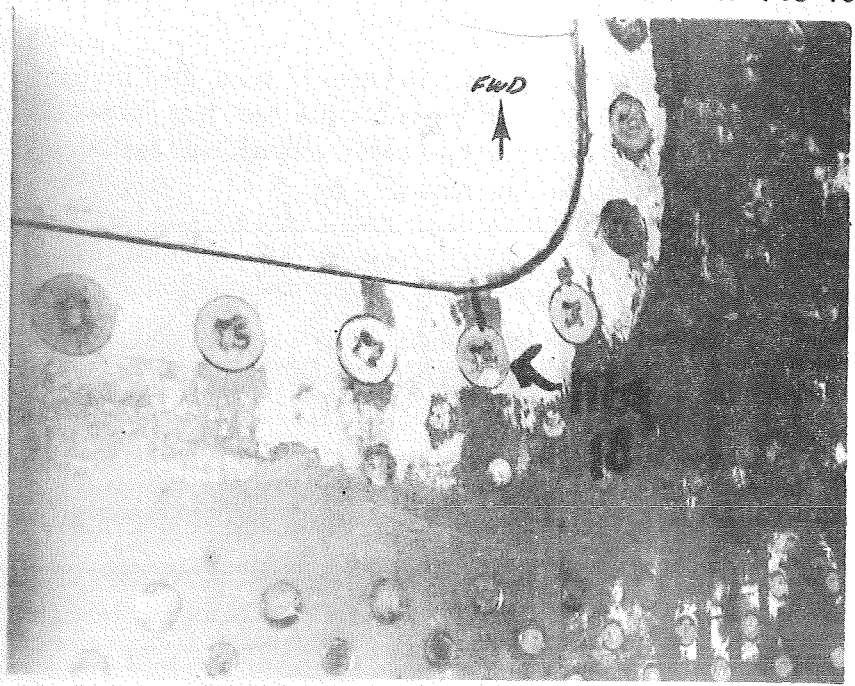


Figure 13-18 WS 112.8R, stringer 19 area, aft inboard corner of lower surface fuel bag access door, prior to testing, showing item 18.

RK 0735-12

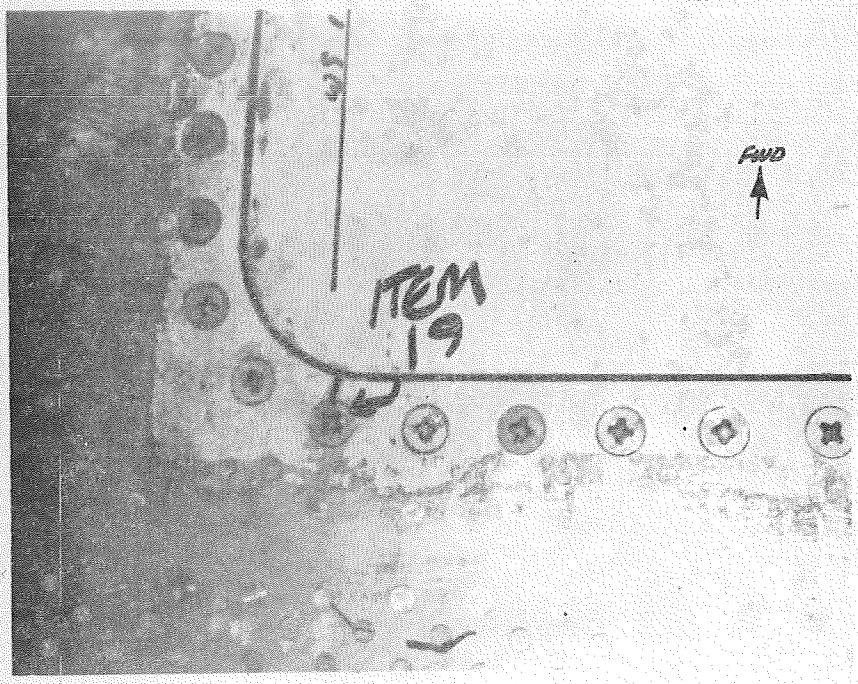


Figure 13-19
WS 128R, stringer 19 area. Aft outboard corner of lower surface fuel bag access door, prior to testing, showing item 19.

RK 0898-9

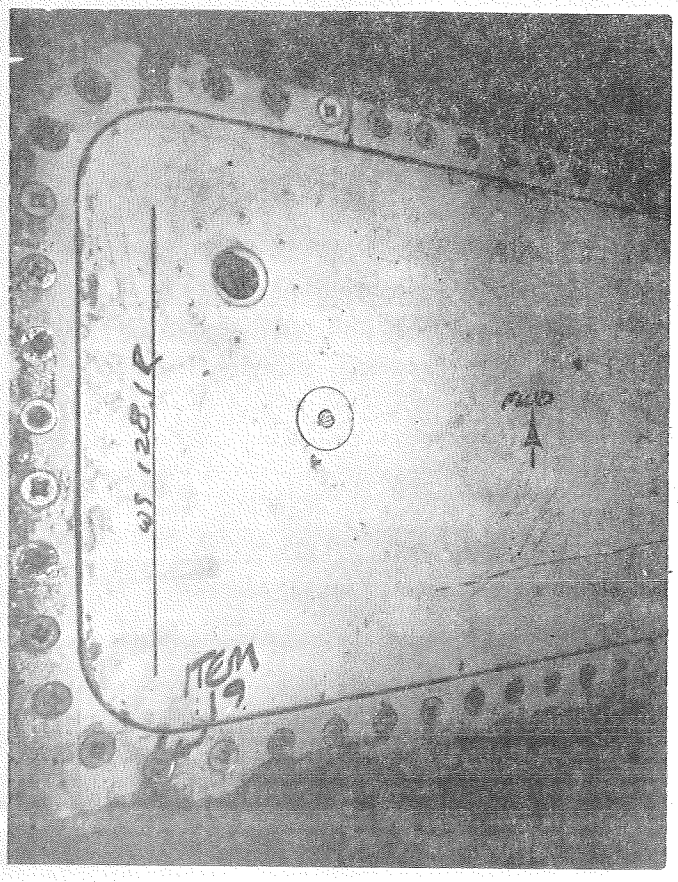


Figure 13-20
WS 128R, stringer 17 - 19 area, lower surface access door, showing item 19.

RK 0735-11

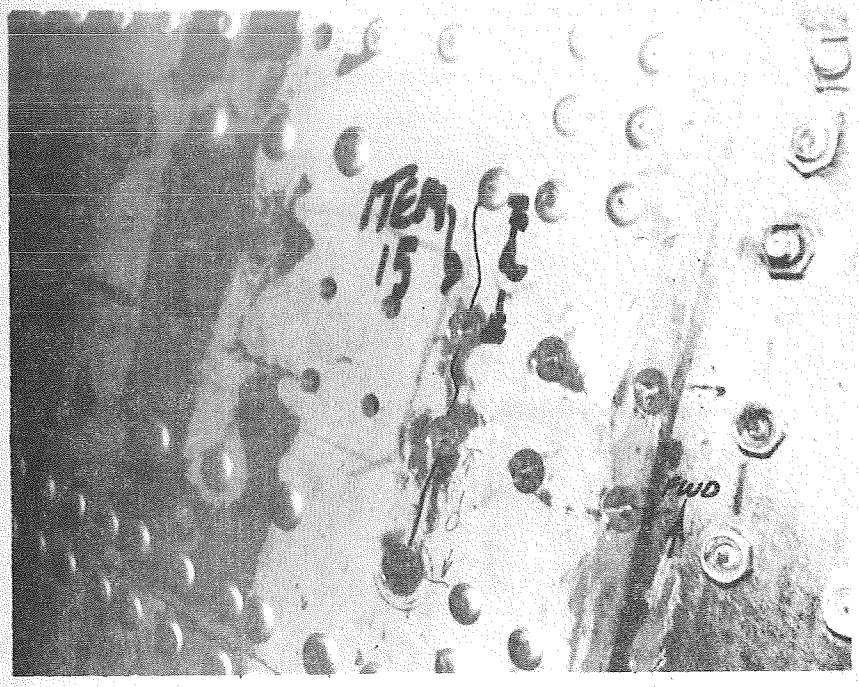
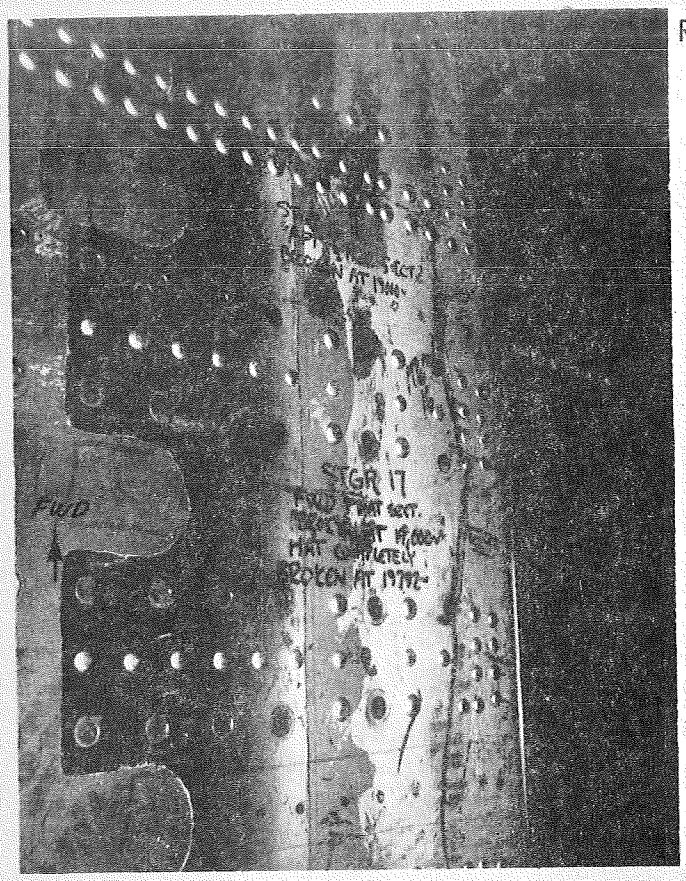


Figure 13-21

WS 58R, stringer 15 area, prior to testing showing item 15.

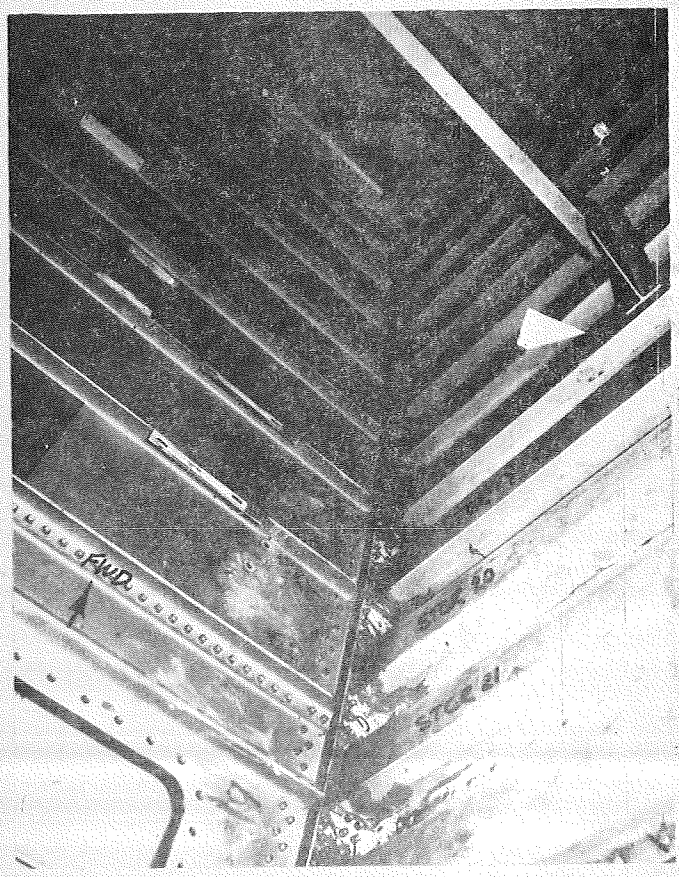
RK 0735-10

Figure 13-22
WS 58R, stringer 16 & 17 area, prior to testing, showing item 16.



RK 0898-22

Figure 13-23
WS 58R, stringers 15 thru 18 area subsequent to crack propagation testing showing items 15 & 16.



RK 0898-23

Figure 13-24
Internal view WS 58R, stringers 12 thru 22, subsequent to crack propagation testing. Showing damage to stringers 16 & 17.

RK 0884-5

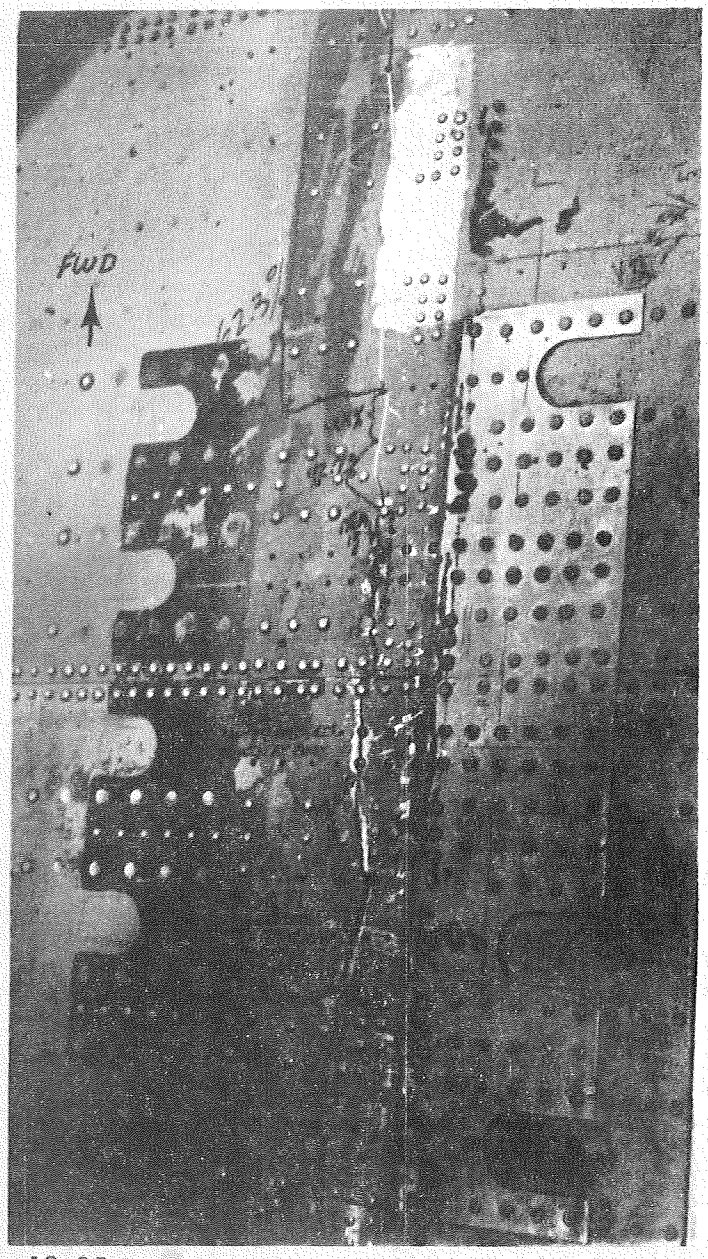


Figure 13-25 WS 58R, stringers 12 thru 18 area, subsequent to failure showing items 15 & 16.

RK 0884-6

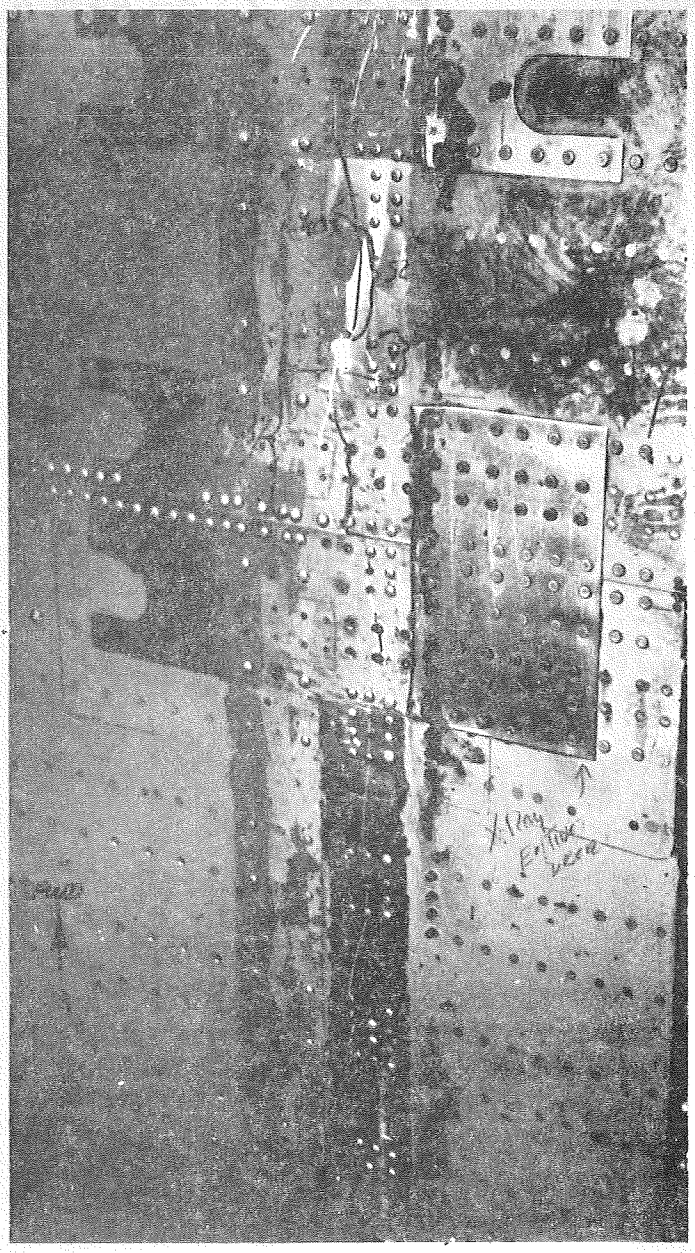
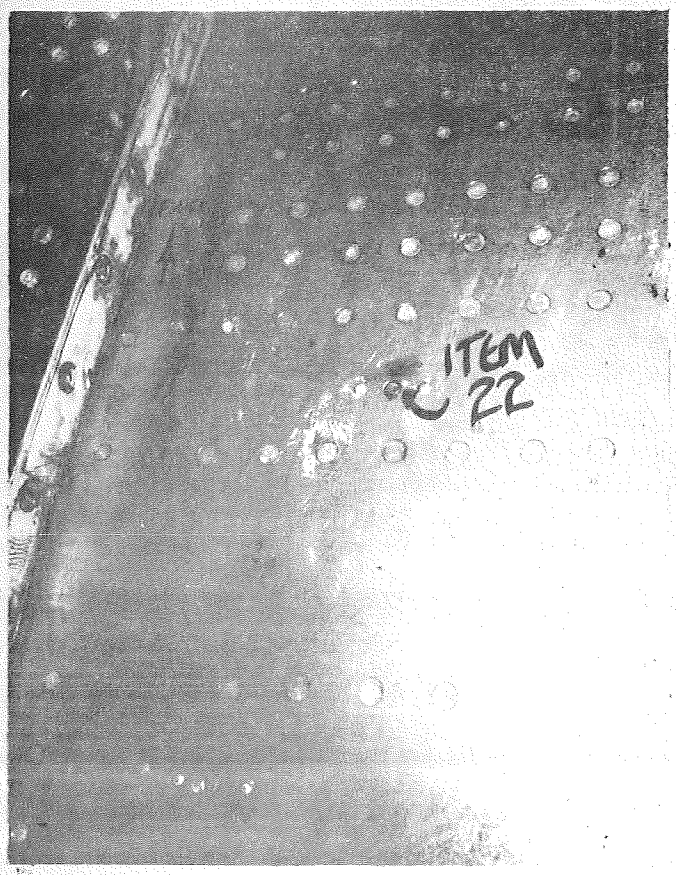


Figure 13-26 WS 58R, stringers 19 thru 24 area, subsequent to failure, showing item 16.

RK 0898-7

Figure 13-27
WS 170R, stringers 12, 13 & 14 area, subsequent to crack propagation testing showing items 32, 36 & 38.



RK 0898-6

Figure 13-28
WS 170R, stringers 16 & 17 area, subsequent to crack propagation testing, showing item 22.

RK 0898-10

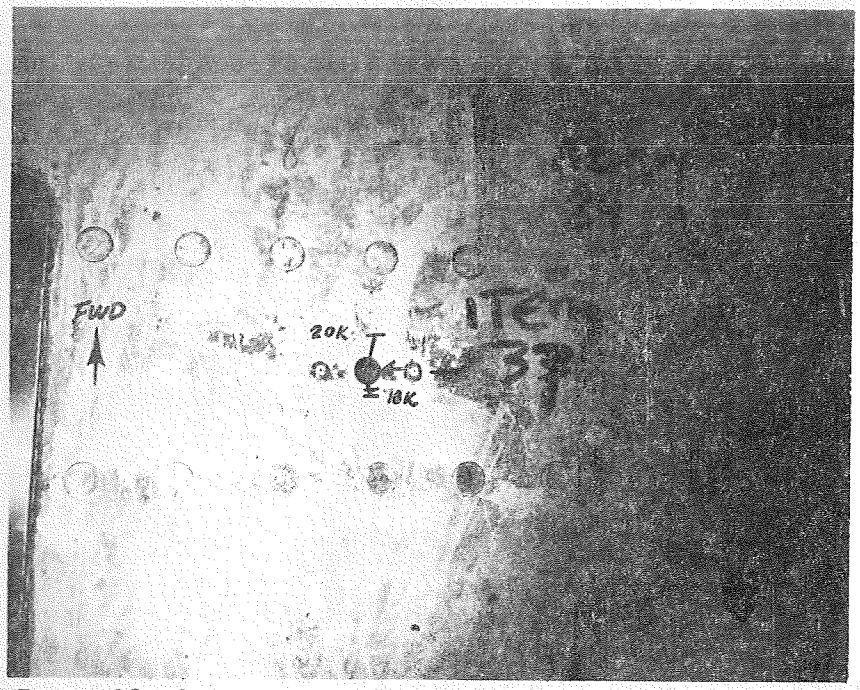


Figure 13-29 WS 170R, stringers 18 & 19 area, subsequent to crack propagation testing showing items 24 & 37.

RK 0898-8

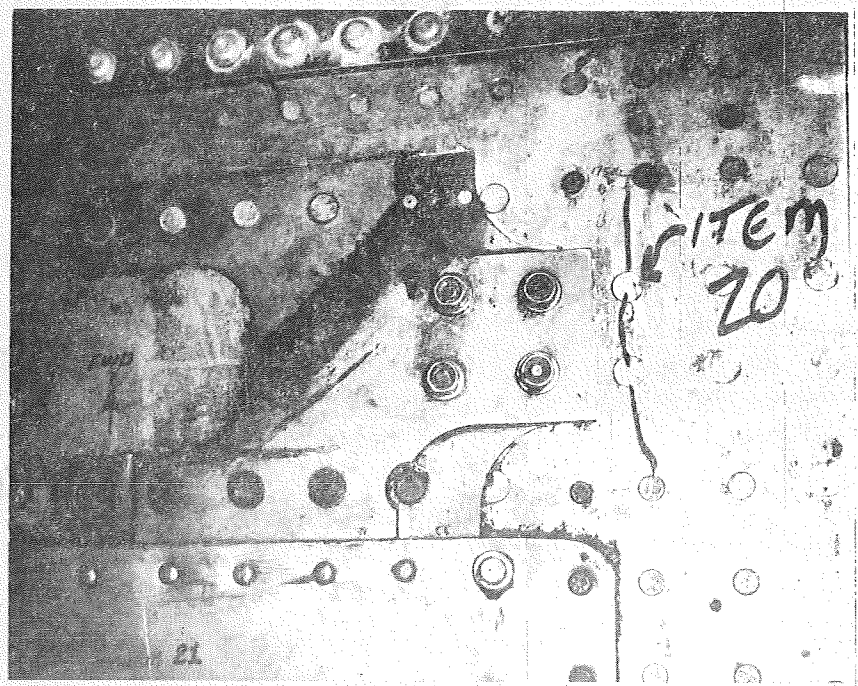


Figure 13-30 WS 176, 186R stringer 21 area subsequent to crack propagation testing showing items 20 & 21.

RK 0884-1

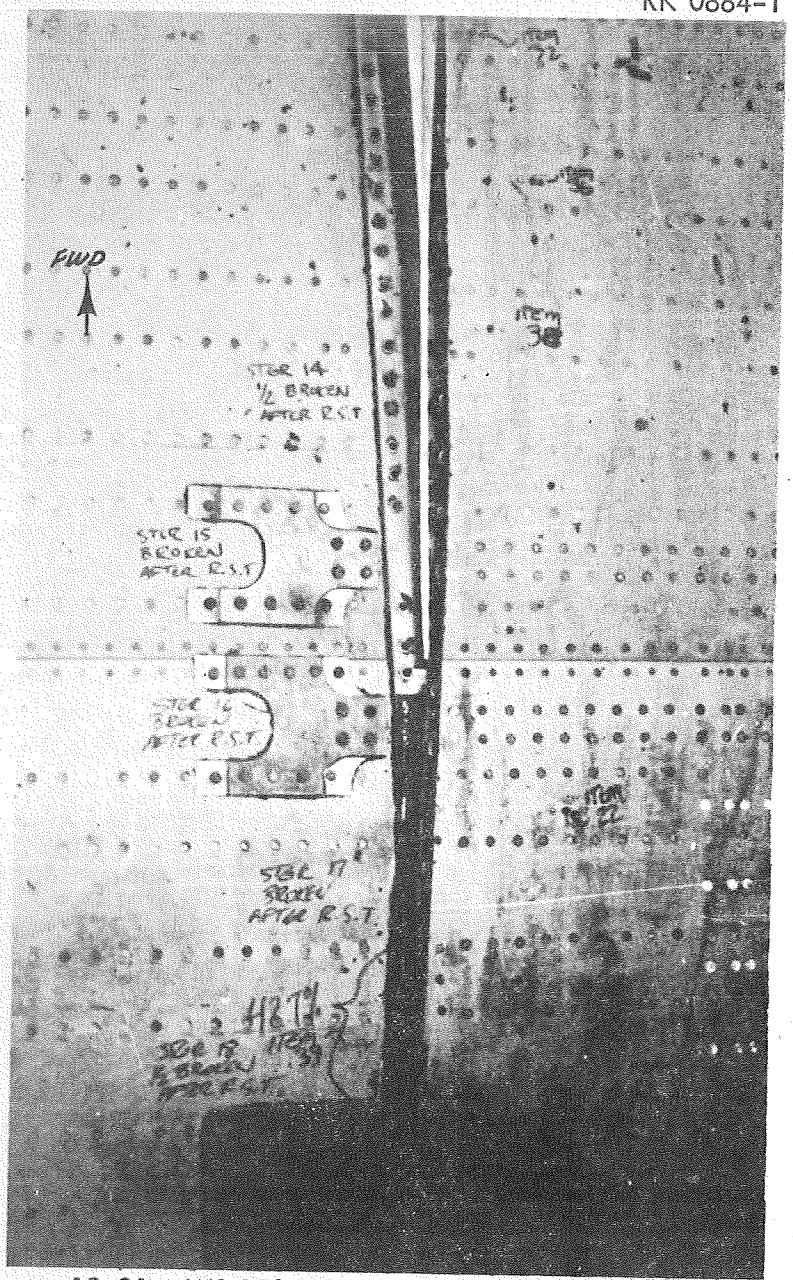


Figure 13-31 WS 170 - 180 R area, front beam to stringer 19, subsequent to failure, showing items 32, 36, 38, 22, 39, 24, & 37.

RK 0884-2

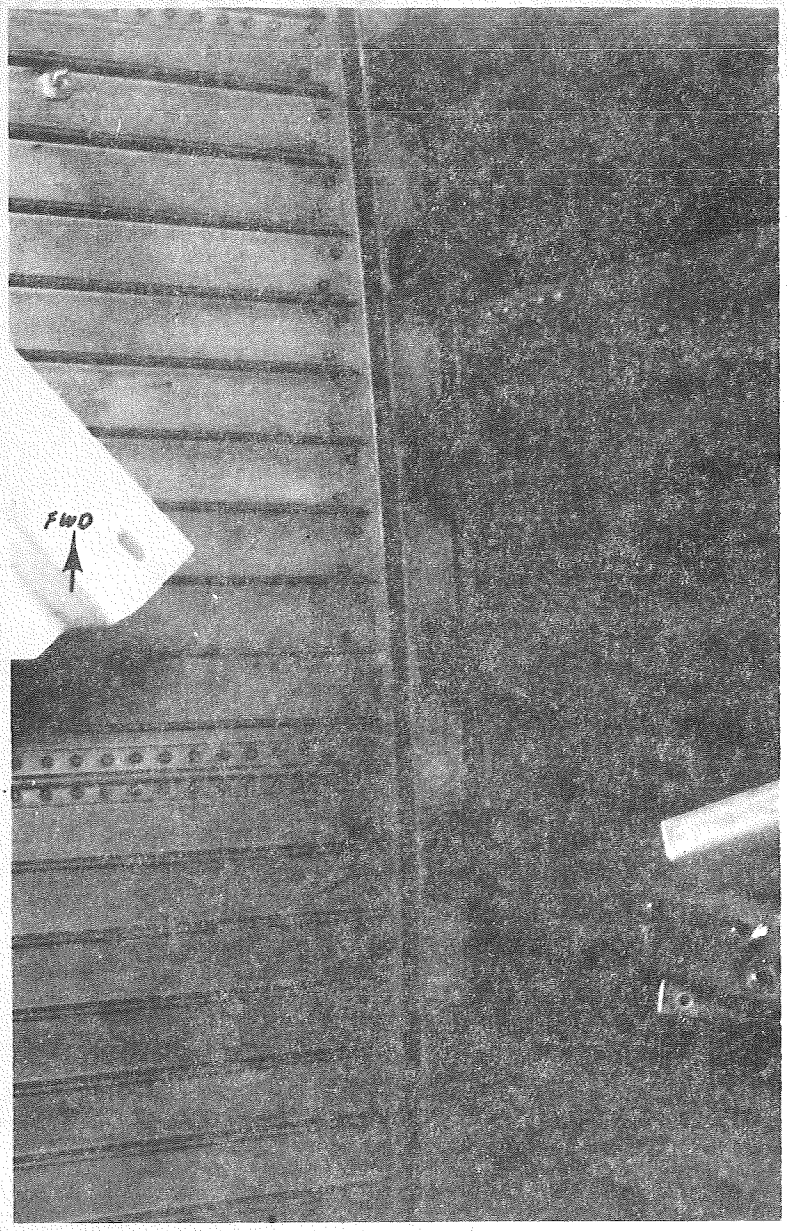


Figure 13-32 Internal view of WS 185-170R, stringers 14 thru 20 area subsequent to failure showing broken 'hat' sections.

PART 14

Specimen 14

CRACK PROPAGATION & RESIDUAL STRENGTH

TEST RESULTS

TEST RESULTS

This was the second commercial C-130 airplane wing box tested, and also had accumulated approximately 13,000 flight hours. The general appearance of the wing box, extent of initial cracking, and the number of large repair patches visible indicated more evident fatigue damage on this wing than on most of the other test specimens. Multiple stainless steel repair patches had been installed by the Air Line operation. X-ray examination of reported cracks under repair patches revealed a 5.3-inch crack (number 24) at WS 59R, so two large patches were removed at this station to permit crack growth during testing without inhibition by repair patches. Crack terminals were at the edge of a panel and in a fastener hole (Figure 14-7). A total of 30 individually identified cracks were observed and recorded at the start of testing.

Initial cyclic load level was set at 45% Limit Load. At the end of 6297 cycles, sonic reports from the specimen were heard and the cycling was halted for immediate specimen inspection. The following damage was detected:

Some segments of a large crack running irregularly under the engine thrust angle at WS 179R were visible (Figures 14-1 and 14-3). Internal inspection disclosed a 36-inch long skin crack under the engine thrust angle and disclosed that Stringers #15, #16, #17, and #18 were broken at this same location. Elsewhere, other cracks which had progressed gradually during test cycling included a 0.1-inch crack (number 23) at WS 59R which had grown to 4.4 inches (Figures 14-5 and 14-6) and several small cracks at WS 168L which had coalesced into a single (number 15) 22-inch crack (Figures 14-9, 14-10, and 14-11). The original 5.3-inch crack at WS 58R had grown only to the next fastener hole and a total length of 5.9 inches (Figure 14-8). Cyclic load level was halved to 22-1/2% Limit Load to reduce the rate of cyclic damage in an attempt to provide further useful crack propagation data despite the large loss of cross-sectional area at WS 179R, and also to preserve some load-carrying capacity for the subsequent static residual strength test.

At 6297 cycles, a total of 60 independently identified cracks was observed. During the 10,000 additional cycles at 22-1/2% Limit Load, another 20 cracks were initiated; none of the larger cracks propagated appreciably more. Failure during static residual strength test occurred at 59% Limit Load at WS 179R (Figures 14-2 and 14-4). Prior to failure, only one sonic report of consequence was detected, when the 22-inch crack at WS 168L jumped 1 inch to the next fastener hole (at 50% Limit Load).

The largest initial crack detected (5.3 inches at WS 59R) never grew to more than 5.9 inches, which was reached after the first 4500 cycles when it advanced to the next fastener hole. Two large steel patches had been removed to permit unrestrained propagation of this crack. In contrast, there was no report of cracking under the engine thrust angle at WS 179R prior to testing, although an aged crack in the blind region between the thrust angle and Stringer #16 was found during post-failure examination of the fractured surface. This is located in the view of the failure surface in Figure 14-4, immediately to the right

TEST RESULTS (Continued)

of the repair patch over Stringer #16. This initial crack extended approximately 1" forward into panel #1 and 6" aft into panel #2. Miscellaneous cracking occurred at WS 179R (Figures 14-12 and 14-13); WS 171 - 173R (Figure 14-19); WS 175L - 179L (Figures 14-14 and 14-15); WS 213.7L (Figures 14-16 and 14-17); and at the WS 112.8L access door corner (Figure 18).

Table I-14 gives a summary of major fatigue damage accumulated by the specimen and Table II-14 gives a summary of minor damage.

TABLE I-14

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 14

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 45% Limit Load — Number of Cycles (d)															Phase 2 Max. Load = 22.5% — Cumulative Cycles (d)		After Failure at 59% Limit Load
					0	500	2000	2500	3000	3500	4000	4500	5000	5500	5650	5685	5950	6022	6297	6500	16,297	
13	The crack originated in the edge of the last fastener hole common to the internal reinforcing doubler of the access door cutout and the skin	168L	2.8" aft of the stringer 16 ℄	14-9	0.2	0.2	0.3	—	—	—	—	—	—	—	0.3							
			a		0.2	0.2	0.3	—	—	—	—	—	—	—	0.3							
			b																			
32	Same as crack 16		1.8" fwd of the stringer 17 ℄	14-10				0.1	—	—	0.1	0.2	—	0.2								
14	Same as crack 13		The stringer 17 ℄	14-9	0.1	—	—	—	—	0.1	0.2	0.6	0.6									
			a		0.1	—	—	—	—	0.1	0.2	0.2	0.2									
			b									0.1	0.1									
35	Same as crack 16		1.8" aft of the stringer 17 ℄	14-10							0.1											
			a																			
			b								0.1											
15	Same as crack 13		2.8" aft of the stringer 17 ℄	14-9	0.1	—	0.1	0.2	—	0.2	4.6	5.8	6.3	10.0	11.2	16.3	19.9	20.4	22.7	—	22.7	23.7
			a		0.1	—	0.1	0.2	—	0.2	0.4 → 3.9	1.2 → 4.3(c)	1.7 → 4.3	3.6 → 6.0	4.3 → 9.5	6.4	7.5	8.0	8.9(c)	—	—	8.9
			b													12.1(c)		13.5	—	—	13.5	14.5
16	Same as crack 13 except the fastener hole was also common to the stringer flange		1.8" fwd of the stringer 18 ℄	14-9	0.1	—	0.1	0.2	0.7	0.9												
			a		0.1	—	0.1	0.2	0.2	0.3												
			b						0.1	0.2												
17	Same as crack 13		The stringer 18 ℄	14-9	1.4	1.4	1.6	1.7	1.9	2.1												
			a		0.6	0.6	0.7	0.7	0.8	0.9												
			b		0.5	0.5	0.6	0.7	0.7	0.8												
18	Same as crack 13		2.8" fwd of the stringer 19 ℄		0.1	0.1	0.2	—	—	0.2	0.3	—	0.3									
			a		0.1	0.1	0.2	—	—	0.2	0.3	—	0.3									
			b																			
19	Same as crack 13	168L	The stringer 19 ℄	14-9	0.8	—	0.8	1.0	1.0	1.1	1.2	—	1.2	1.5	1.5							
			a		0.2	—	0.2	0.3	0.3	0.4	—	—	0.4	0.6	0.6							
			b		0.2	—	0.2	0.3	0.3	0.4	—	—	0.4	0.6	0.6							
6	The crack originated in the edge of a fastener hole common to the skin and the stringer flange	172.7L	1.8" fwd of the stringer 12 ℄		0.6	—	0.6	0.7	0.8	0.9	0.9	1.0	1.0	1.1	—	—	—	—	1.1	1.2	—	1.2
			a		0.1	—	0.1	0.2	0.2	0.3	—	—	0.3	0.3	—	—	—	—	0.3	0.4	—	0.4
			b		0.1	—	0.1	0.2	0.2	0.3	—	—	0.3	0.4	—	—	—	—	0.4	0.5	—	0.5

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-14 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 14

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 45% Limit Load — Number of Cycles (d): 0	500	2000	2500	3000	3500	4000	4500	5000	5500	5650	5685	5950	6022	6297	Phase 2 Max. Load = 22.5% — Cumulative Cycles (d): 6500	16,297	After Failure at 59% Limit Load
5	The crack originated in the edge of a drain hole in the skin	173L	2.5" aft of fwd edge of panel #1		0.7				0.7	0.9	1.0								1.0	1.3		1.3
			a		0.1				0.1	0.3	0.3								0.3	0.4		0.4
			b		0.2				0.2	0.4	0.4								0.4	0.6		0.6
4	The crack originated in the edge of a fastener hole common to the skin and the forward spar cap	173.5L	0.7" aft of fwd edge of panel #1		1.5	1.5	2.1	2.2	2.3	2.4									2.4	2.6		2.6
			a		0.1	0.1	0.5	0.5(b)		0.5									0.5	0.5		0.5
			b		1.1	1.1	1.3	1.4	1.4	1.5									1.5	1.7		1.7
9	The crack originated in the edge of a fastener hole near the end of the internal reinforcing beam of the access door cutout	176L	0.5" fwd of the stringer 16 ℄		0.5						0.5	0.6							0.6	1.1	1.1	1.3
			a		0.1														0.1	0.2		0.2
			b		0.1																0.1	0.2
			℄																	0.1	0.1	0.1
10	Same as crack 9.	179L	0.5" fwd & aft of the stringer 21 ℄	14-14 14-15																0.1		0.1
			a		0.2						0.2	0.6	0.7							0.7	1.0	1.0
			b									0.1										0.1
			c		0.1							0.1	0.2							0.2	0.3	0.3
			d		0.1															0.1	0.3	0.3
40	Same as crack 31	182L	3.0" aft of the stringer 15 ℄										0.6						0.6	1.1		1.1
			d										0.6(c)						0.6	1.1	1.1	
31	The crack originated under an external sling shot doubler repair at the end of the internal reinforcing beam of the access door cutout	182L	3.0" fwd of the stringer 16 ℄			0.4(h) (b)				0.4	5.2(j)	5.5	5.9	6.2				6.2	6.4			6.4

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE I-14 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 14

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 45% Limit Load — Number of Cycles (d): 0	500	2000	2500	3000	3500	4000	4500	5000	5500	5650	5685	5950	6022	6297	Phase 2 Max. Load = 22.5% — Cumulative Cycles (d): 6500	16,297	After Failure at 59% Limit Load
3	The crack originated in the edge of the last fastener inboard common to the wing joint fitting and skin	213.7L	2.8" aft of the stringer 16 ℄	14-16 14-17	0.8	0.8	1.0	1.0	1.2	1.3	1.3	1.3	1.5						1.5	2.1	2.1	2.1
			a		0.1	0.1	0.2	0.2	0.3	0.3	0.3	0.3	0.4						0.4	0.6	0.6	0.6
			b		0.3	0.3	0.5	0.5	0.6	0.6	0.6	0.6	0.7(c)						0.7	1.2	1.2	1.2
36	Same as crack 3 except the fastener hole was also common to the stringer flange tie-thru angle	213.7L	1.8" fwd of the stringer 17 ℄									0.2							0.2			
			a									0.2							0.2			
23	The crack originated in the edge of the last fasteners inboard common to the skin and the W.S. 120 door reinforcement	59R	0.5" aft of the stringer 16 ℄	14-5 14-6	0.1	0.8	0.8	1.1	1.3	2.4	3.2	3.8	4.2	4.4					4.4	5.0	5.0	5.4
			a							0.2	0.6	1.0(c)									1.0	1.4(b)
			b			0.1			0.1													
			c		0.1	0.2	0.2	0.5	0.7	0.8	1.1	1.4	1.7	1.9(c)					1.9	3.5(c)		3.5
			d																			
24	Same as crack 23, except the crack was measured from the aft edge of panel #2	59R	The stringer 20 ℄	14-7 14-8	5.3(c) (b)						5.3	5.9(c)										5.9
27	The crack originated in the edge of fasteners common to the skin and the W.S. 120 door reinforcement beam in the last outboard fasteners not common to the repair	179R	0.5" fwd & aft of the stringer 21 ℄	14-12 14-13	2.5					2.5	2.8	2.9	3.0	3.1				3.1	3.3		3.3	
			a																			
			b		0.6					0.6	0.7	0.9						0.9	1.1			
			c																			
			d		0.6					0.6	0.7	0.7	0.9	1.0(c)							1.0	
58	The crack originated in the edge of a fastener common to the skin and a stringer flange, and grew so as to 'disappear' under the engine drag angle	182R	1.8" aft of the stringer 14 ℄	14-1 14-2 14-3														5.4	18.8(g)		18.8	Complete Chordwise Failure
			a															3.2	11.2		11.2	
			b															1.9	7.3		7.3	

NOTE: Footnotes found in Table III of the General Section, Part G.

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TABLE I-14 (Cont.)

SUMMARY OF MAJOR CRACK LENGTHS, SPECIMEN 14

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta. In.	Crack Location	Fig. No.	Upbending, Phase 1 Cyclic Test, R = 0.1 Max. Load = 45% Limit Load — Number of Cycles (d): 0	500	2000	2500	3000	3500	4000	4500	5000	5500	5650	5685	5950	6022	6297	Phase 2 Max. Load = 22.5% — Cumulative Cycles (d): 6500	16,297	After Failure at 59% Limit Load
60	The crack originated underneath the engine drag angle at a point invisible from either side	180R	Aft from fwd edge of panel #2																17.5(g)	17.5	Part of Complete Chordwise Failure
56	Same as crack 31	182L	3.0" fwd of the stringer 21 ℄	14-15									0.7	1.2	—	—	—	—	—	—	1.2
			a										0.7(c)	1.2(b)	—	—	—	—	—	—	1.2
			b																		

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-14

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 14

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta., In.	Crack Location	Figure No.	No. of Cycles When Discovered	Length When Discovered, In.	(d) Length After 16,297 Cycles	Length After Failure at 59% Limit Load
1	The crack originated at fwd inbrd corner of wing joint bolt installation cutout	214.7L	1.0" fwd of the stringer 12 ℄		0	0.2	0.4	0.4
2	Same as crack 1	214.7L	1.0" fwd of the stringer 14 ℄		0	0.3	0.7(c)	0.7(c)
7	The crack originated at the second fastener aft common to panel #1 and the forward lower beam cap	174.3L	0.5" aft of the forward edge of panel #1		0	0.1	0.1	0.1
			a			-	-	-
			b			0.1	0.1	0.1
8	The crack originated in the edge of fastener holes common to the skin and the W.S. 120L door cutout reinforcement beam	177L	0.5" fwd & aft of the stringer 15 ℄		0	0.2	0.7	0.7
			a				0.1	0.1
			b			0.1	0.2	0.2
			c			0.1	0.1	0.1
11	Same as 8	178L	0.5" fwd & aft of the stringer 21 ℄	14-14	0	0.1	0.2	0.2
			c			0.1	0.2	0.2
			d			-	-	-
12	Same as 8	175L	0.5" fwd & aft of the stringer 21 ℄	14-14	0	0.1	0.1	0.1
			d			0.1	0.1	0.1
20	The crack originated in the edge of the last fastener hole common to the internal reinforcing doubler of the access door cutout. and the skin	168L	2.8" aft of the stringer 19 ℄	14-10	0	0.1	0.7	0.7
			a			0.1	0.3	0.3
			b			-	0.1	0.1
21	The crack originated in the edge of a door attachment fastener hole	112.8L	0.8" fwd of the stringer 17 ℄	14-18	0	0.3	0.3	0.3

NOTE: Footnotes found in Table III of the General Section, Part C.

TABLE II-14

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 14

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta., In.	Crack Location	Figure No.	No. of Cycles When Discovered	Length When Discovered, In.	(d) Length After 16,297 Cycles	Length After Failure at 59% Limit Load
22	The crack originated in the edge of a fastener hole common to panel #1 and the aft lower beam cap	58.8L	1.1" fwd of aft eage panel #3		0	0.3	0.3	0.3
			a			0.3	0.3	0.3
			b			-	-	-
25	The crack originated in the edge of a drain hole in the skin	171R	2.8" fwd of the stringer 12 ℄	14-19	0	0.1	0.1	0.3
			a			0.1	0.1	0.3
			b			-	-	-
26	Same as 25	175R	2.8" fwd of the stringer 20 ℄	14-12	0	0.1	0.3	0.3
			a			0.1	0.3	0.3
			b			-	-	-
28	Same as crack 8	175R	0.5" fwd & aft of the stringer 21 ℄	14-13	0	0.2	0.2	0.2
			c			-	-	-
			d			0.2	0.2	0.2
29	Same as crack 1	214.7R	1.0" fwd of the stringer 13 ℄		0	0.3	0.6	0.6
30	Same as crack 1	214.7R	1.0" fwd of the stringer 14 ℄		0	0.1	0.2	0.2
33	The crack originated in the edge of a fastener hole common to the skin and a stringer flange (adjacent to a repair)	187R	1.8" fwd of the stringer 15 ℄		3000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-
34	Same as crack 8	177R	0.5" fwd & aft of the stringer 21 ℄	14-13	4000	0.1	0.1	0.1
			c			-	-	-
			d			0.1	0.1	0.1
37	The crack originated in the edge of a fastener hole common to the skin, a stringer flange, and the W.S. 120 door cutout reinforcement doubler	167L	1.8" fwd of the stringer 19 ℄		5000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-14

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 14

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta., In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered, In.	(d) Length After 16,297 Cycles	Length After Failure at 59% Limit Load
38	Same as crack 37	167L	1.8" aft of the stringer 18 ℄		5000	0.3	0.3	0.3
			a			0.3	0.3	0.3
			b			-	-	-
39	The crack originated in the edge of a fastener hole common to the skin and a stringer flange	169.5L	1.8" aft of the stringer 16 ℄		5000	0.1	0.2	0.2
			a			0.1	0.2	0.2
			b			-	-	-
41	Same as crack 39	191R	1.8" aft of the stringer 14 ℄	14-1	5000	0.3	0.3	0.3
			a			0.3	0.3	0.3
			b			-	-	-
42	Same as crack 39	191R	1.8" aft of the stringer 15 ℄		5000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-
43	Same as crack 39	190R	1.8" aft of the stringer 15 ℄		5000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-
44	Same as crack 39	189R	1.8" aft of the stringer 14 ℄		5000	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
45	Same as crack 39	188R	1.8" aft of the stringer 14 ℄		5000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-
46	Same as crack 39	187R	1.8" aft of the stringer 14 ℄		5000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-
47	Same as crack 39	186R	1.8" aft of the stringer 14 ℄		5000	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
48	Same as crack 39	185R	1.8" aft of the stringer 14 ℄		5000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-
49	Same as crack 39	200R	1.8" aft of the stringer 14 ℄	14-1	5000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-14

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 14

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta., In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered, In.	(d) Length After 16,297 Cycles	Length After Failure at 59% Limit Load
50	Same as crack 39	199R	1.8" aft of the stringer 14 ℄	14-1	5000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-
51	Same as crack 39	196R	1.8" aft of the stringer 14 ℄		5000	0.3	0.3	0.3
			a			0.3	0.3	0.3
			b			-	-	-
52	Same as crack 39	195R	1.8" aft of the stringer 14 ℄		5000	0.2	0.2	0.2
			a			0.2	0.3	0.3
			b			-	-	-
53	Same as crack 39	202R	1.8" aft of the stringer 13 ℄		5000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-
54	Same as crack 39	183R	1.8" aft of the stringer 14 ℄		5000	0.1	0.2	0.2
			a			0.1	0.2	0.2
			b			-	-	-
55	Same as crack 39	181R	1.8" aft of the springer 14 ℄		5000	0.2	0.3	0.3
			a			0.2	0.3	0.3
			b			-	-	-
57	Same as crack 39	193R	1.8" aft of the stringer 14 ℄		5950	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-		
59	Same as crack 39	192R	1.8" aft of the stringer 14 ℄		6022	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
61	Same as crack 39	177R	1.8" aft of the stringer 14 ℄		6500	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
62	Same as crack 39	206R	1.8" aft of the stringer 13 ℄		6500	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
63	Same as crack 39	201R	1.8" aft of the stringer 13 ℄	14-1	6500	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-14

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 14

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta., In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered, In.	(d) Length After 16,297 Cycles	Length After Failure at 59% Limit Load
64	Same as crack 39	196R	1.8" aft of the stringer 13 ℄	14-1	6500	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
65	Same as crack 39	195R	1.8" aft of the stringer 13 ℄		6500	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
66	Same as crack 39	188R	1.8" aft of the stringer 13 ℄		6500	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
67	Same as crack 39	186R	1.8" aft of the stringer 13 ℄		6500	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-
68	Same as crack 39	197R	1.8" aft of the stringer 14 ℄	14-1	6500	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
70	Same as crack 20	168R	At the stringer 18 ℄	14-3	8500	0.6	0.6	0.6
			a			0.1	0.1	0.1
			b			0.1	0.1	0.1
71	The crack originated in the edge of a drain hole in the skin	173R	2.5" aft of fwd edge of panel #1	14-19	8500	0.1	0.1	0.4
			a			0.1	0.1	0.2
			b			-	-	0.1
72	Same as crack 8	177R	0.5" fwd & aft of the stringer 21 ℄		9000	0.1	0.1	0.1
			c			-	-	-
			d			0.1	0.1	0.1
73	Same as crack 39	194R	1.8" aft of the stringer 13 ℄	14-1	9000	0.2	0.2	0.2
			a			0.2	0.2	0.2
			b			-	-	-
74	Same as crack 39	192R	1.8" aft of the stringer 13 ℄	14-1	9000	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-

NOTE: Footnotes found in Table III of the General Section, Part G.

TABLE II-14

SUMMARY OF MINOR CRACK LENGTHS, SPECIMEN 14

LOWER SURFACE SKIN, (a)

Crack No.	Description	Wing Sta., In.	Crack Location	Fig. No.	No. of Cycles When Discovered	Length When Discovered, In.	(d) Length After 16,297 Cycles	Length After Failure at 59% Limit Load
75	Same as crack 8	170L	0.5" fwd & aft of the stringer 16 ℄		14,500	0.1	0.1	0.1
			a			-	-	-
			b			0.1	0.1	0.1
76	Same as crack 8	171L	0.5" fwd & aft of the stringer 16 ℄		14,500	0.5	0.5	0.5
			c			0.1	0.1	0.1
			d			0.1	0.1	0.1
77	Same as crack 8	172L	0.5" fwd & aft of the stringer 16 ℄		14,500	0.1	0.1	0.1
			a			-	-	-
			b			0.1	0.1	0.1
78	Same as crack 8	173L	0.5" fwd & aft of the stringer 16 ℄		14,500	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
79	Same as crack 8	174L	0.5" fwd & aft of the stringer 16 ℄		14,500	0.1	0.1	0.1
			a			0.1	0.1	0.1
			b			-	-	-
80	Same as crack 8	177L	0.5" fwd & aft of the stringer 16 ℄		15,000	0.1	0.1	0.1
			c			0.1	0.1	0.1
			d			-	-	-
81	Same as crack 39	176R	1.8" fwd of the stringer 22 ℄		(i)	0.1	-	0.1
			a			0.1		0.1
			b			-		-
82	Same as crack 33	187.5R	1.8" aft of the stringer 21 ℄		(i)	0.1	-	0.1
						0.1	-	0.1
						-	-	-
83	Same as crack 39	170.5R	1.8" aft of the stringer 13 ℄		(i)	0.6	-	0.6
			a			0.2	-	0.2
			b			0.1	-	0.1

NOTE: Footnotes found in Table III of the General Section, Part G.

RK 0853-5

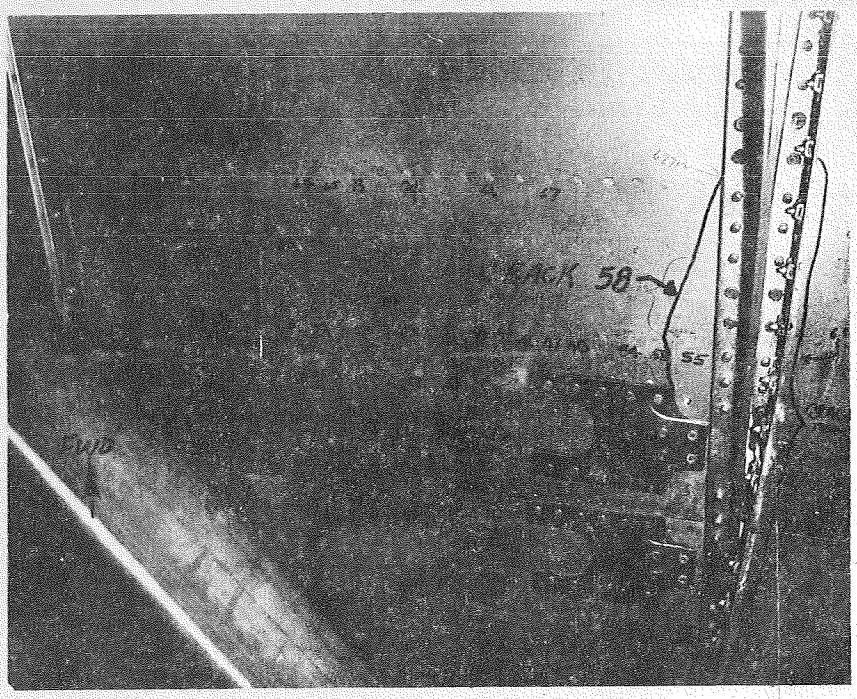


Figure 14-1 WS 180 area of final chordwise failure after completion of the propagation test.

RK 0916-8

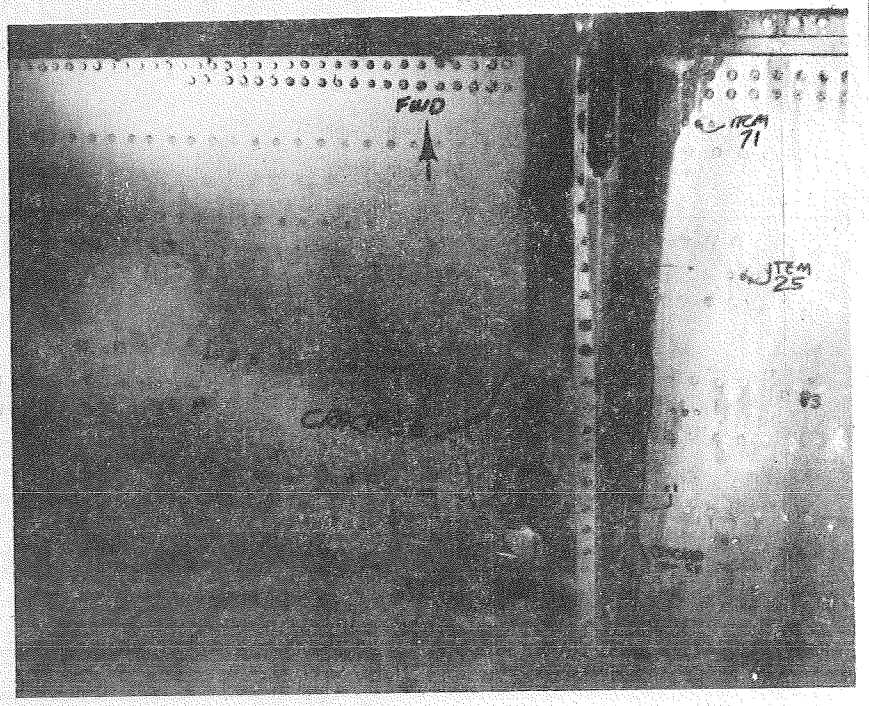


Figure 14-2 Final Failure
Same area as above after final failure at 59% of limit load.

RK 0853-6

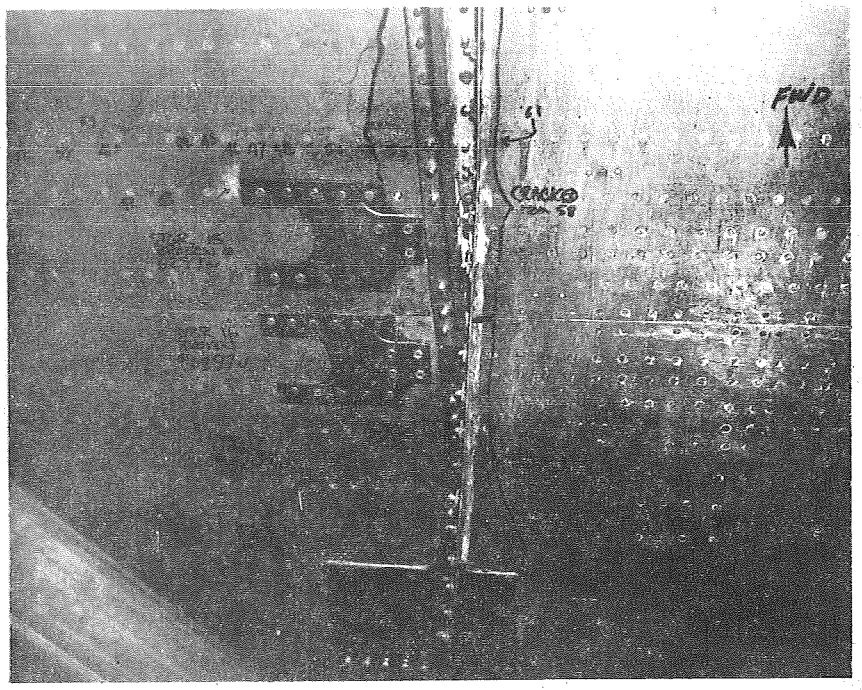


Figure 14-3 WS 180R, area of final chordwise failure after completion of the propagation test.

RK 0916-9

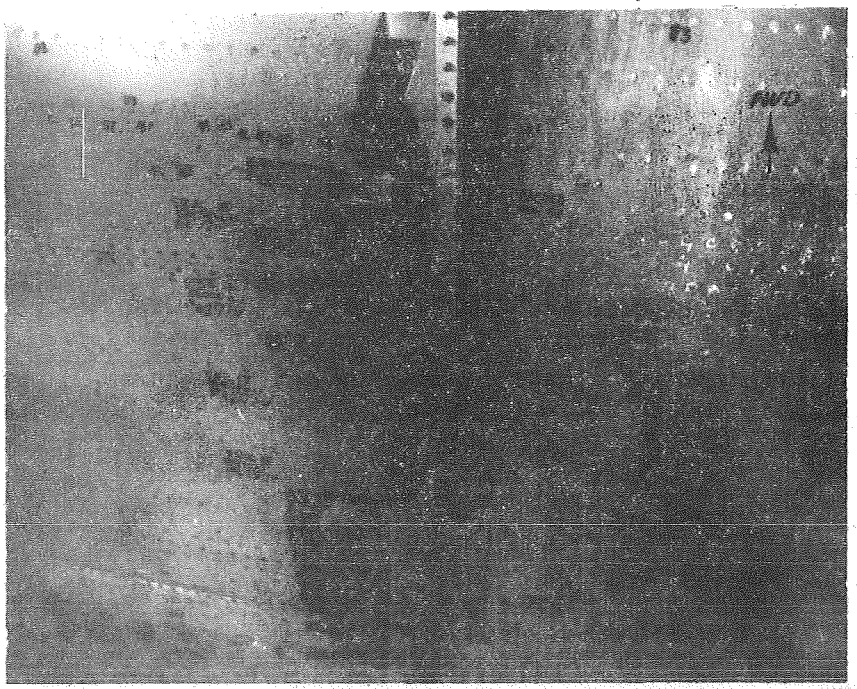


Figure 14-4 Final Failure
Same area as above after final failure at 59% of limit load.

RK 0882-7

Figure 14-5 Initial Damage
Crack 23 was located under stringer 16 at WS 59R. The crack was 0.1 inch long. It originated in a fastener hole common to the skin and the end of the reinforcing beam.

RK 0853-1

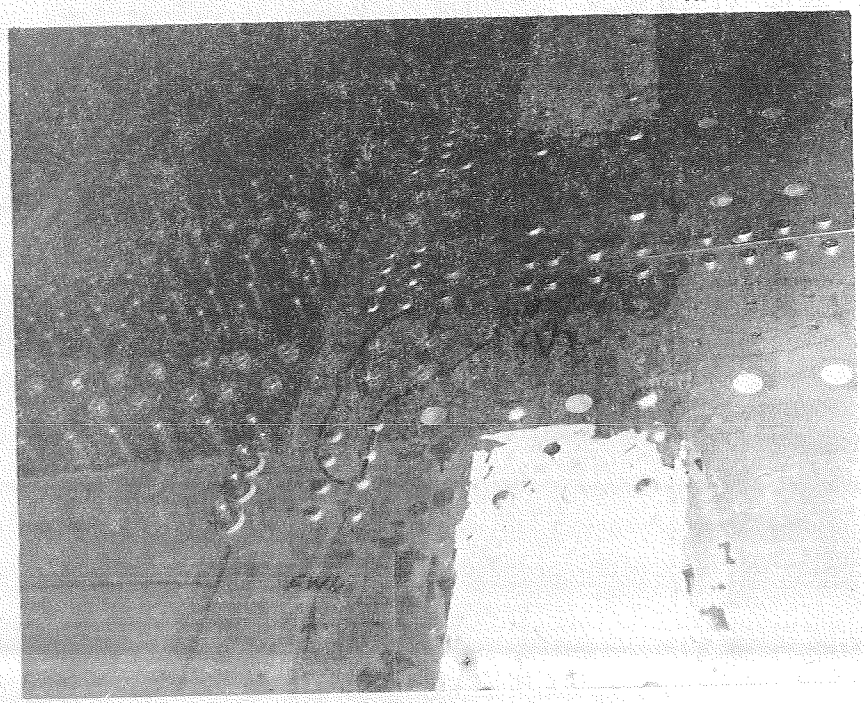


Figure 14-6 Propagation
After completing the propagation test crack 23 had extended to a length of 5 inches.

RK 0882-10

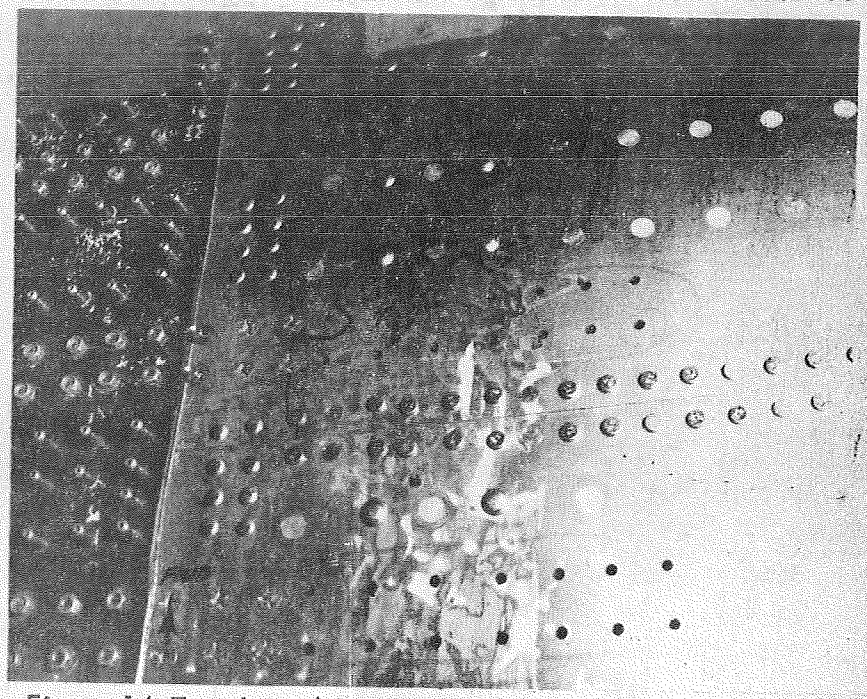


Figure 14-7 Initial Damage
Crack 24, 5.3 inches long, was located under stringer 20 at W.S. 59R.

RK 0853-2.

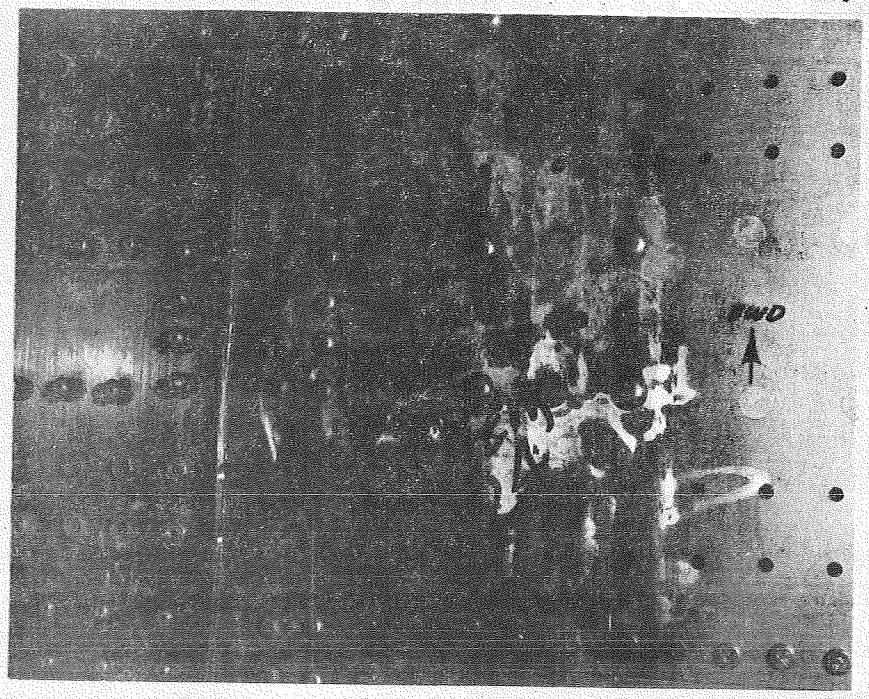
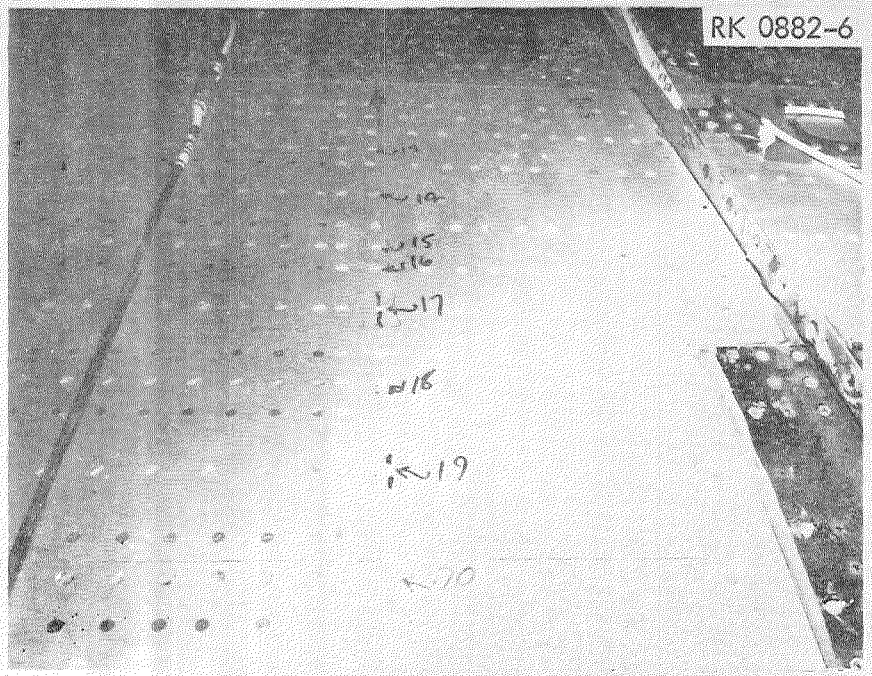
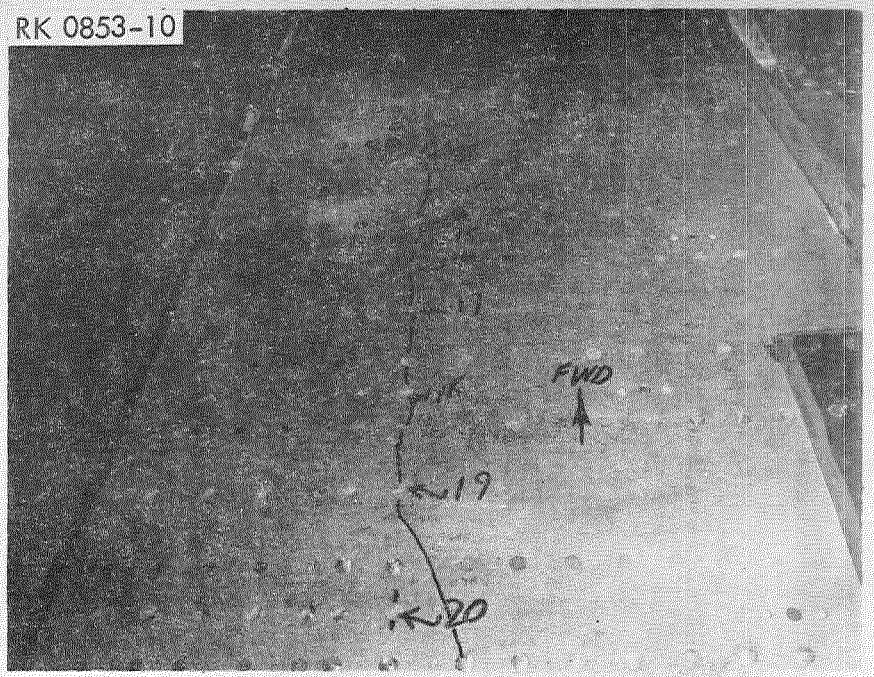


Figure 14-8 Propagation
During crack propagation test of 16,297 cycles, crack 24 extended into adjacent fastener hole.





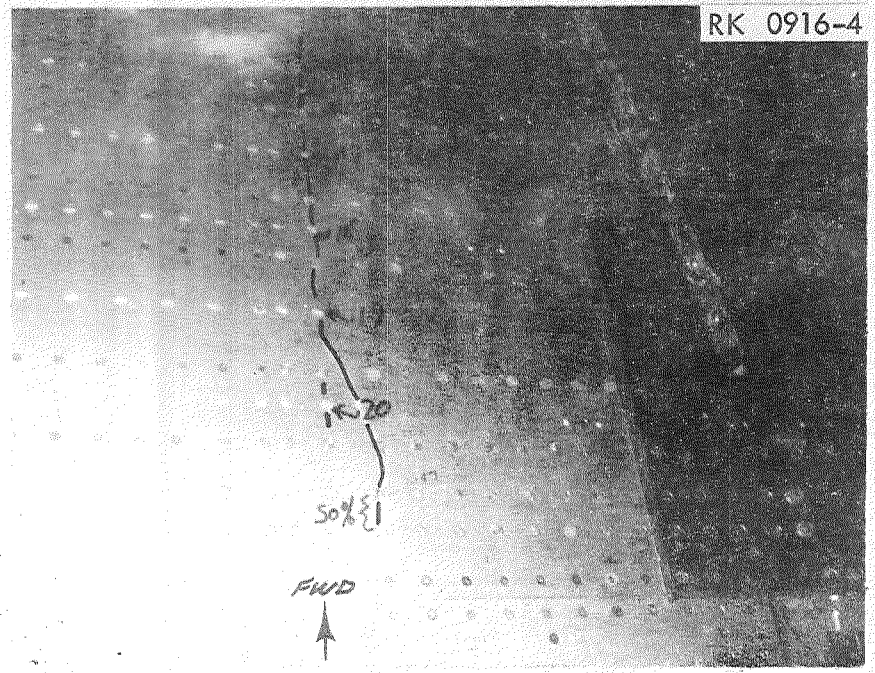


Figure 14-9 (Upper Left)
WS 168L door doubler run out area before testing.

Figure 14-10 (Upper Right)
WS 168L door doubler run out area after propagation testing to 16,297 cycles.

Figure 14-11 (Left)
WS 168L door doubler run out area after testing completed. Note one inch extension found after applying 50% of limit load.

RK 0882-9

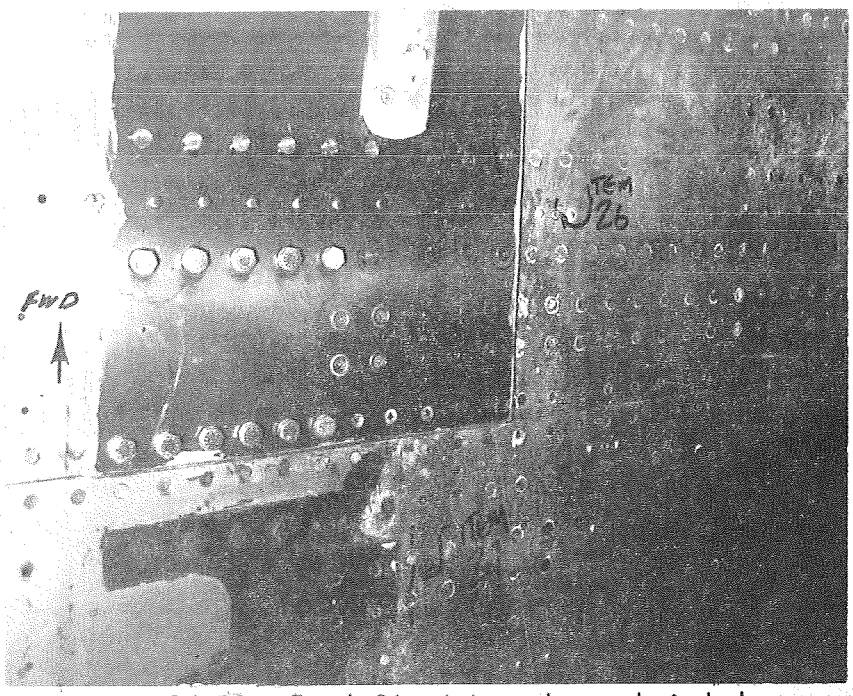


Figure 14-12 Crack 26 originated at a drain hole at WS 175R. Crack 27 originated in the end area of the reinforcing beam under stringer 21 at WS 179R.

RK 0853-14

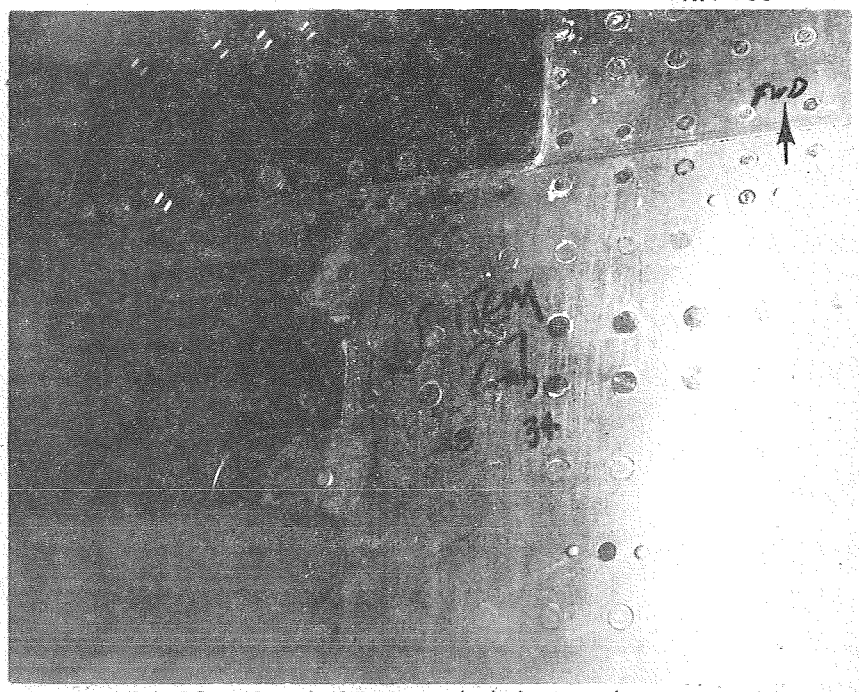


Figure 14-13 Crack 27 extended during the propagation test. Cracks 28 and 34 are located in same area as 27.

RK 0882-1

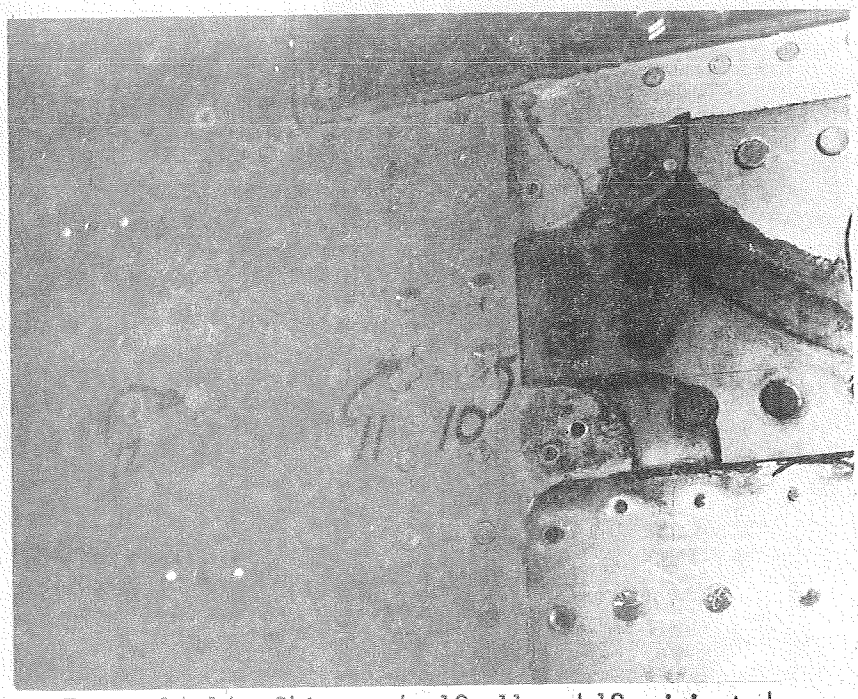


Figure 14-14 Skin cracks 10, 11 and 12 originated in fastener holes common to the reinforcing beam under stringer 21, WS 175L - 179L.

RK 0853-9

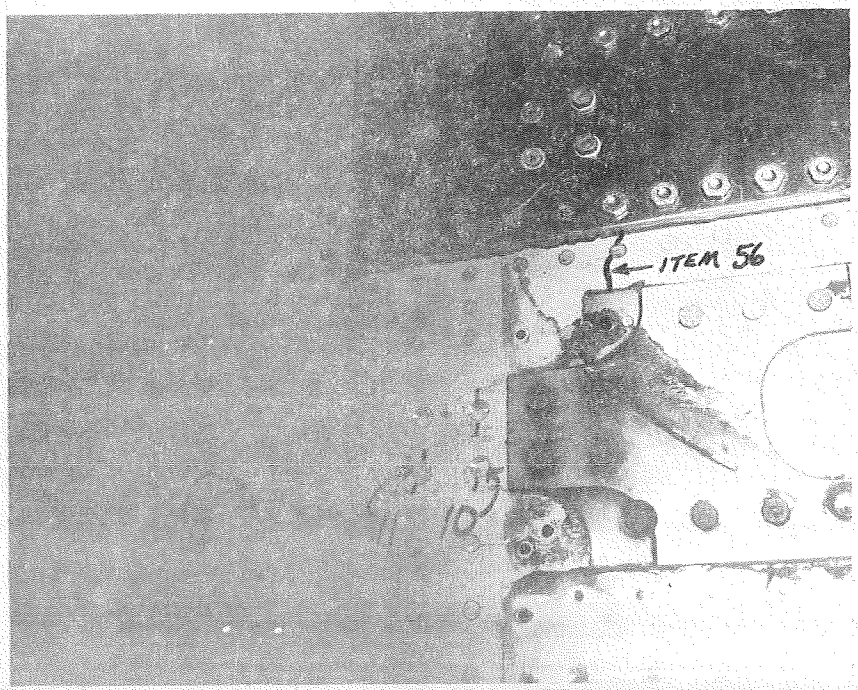


Figure 14-15 Crack 10 had a total length of one inch after the propagation test of 16,297 cycles. Note crack 56.

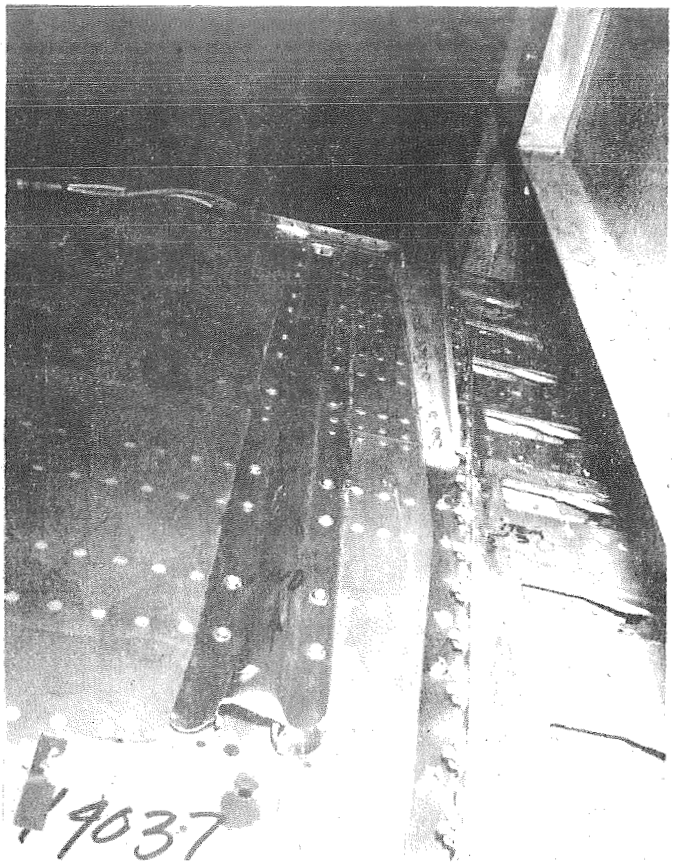


Figure 14-16
Location of crack 3 between stringers 16 and 17 at WS 213.7L

RK 0853-8

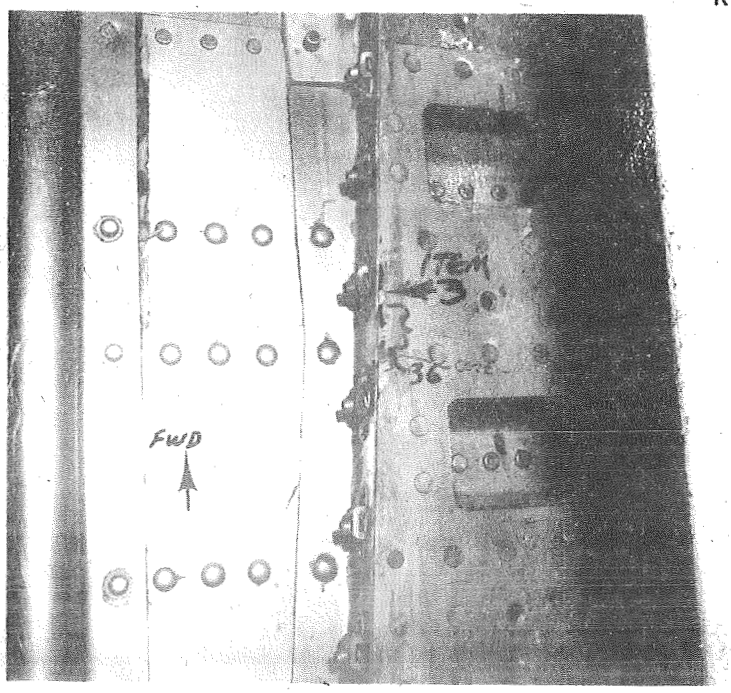


Figure 14-17
Crack 36 originated in the first fastener hole aft of the origin of crack 3. 3 and 36 joined at 6500 cycles.

RK 0882-5

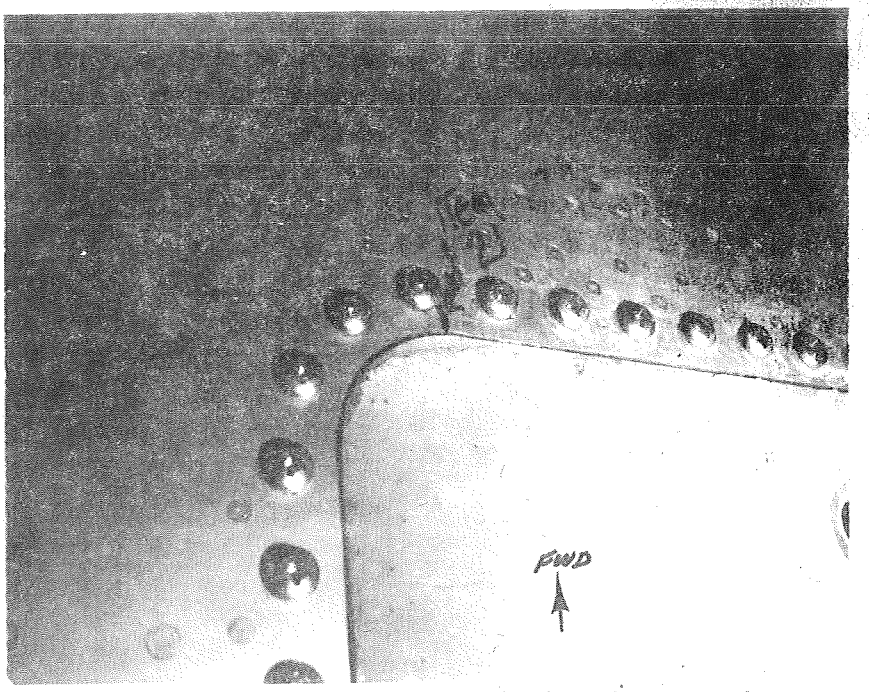


Figure 14-18 Crack 21 in fwd inboard corner of access door cutout.

RK 0853-13

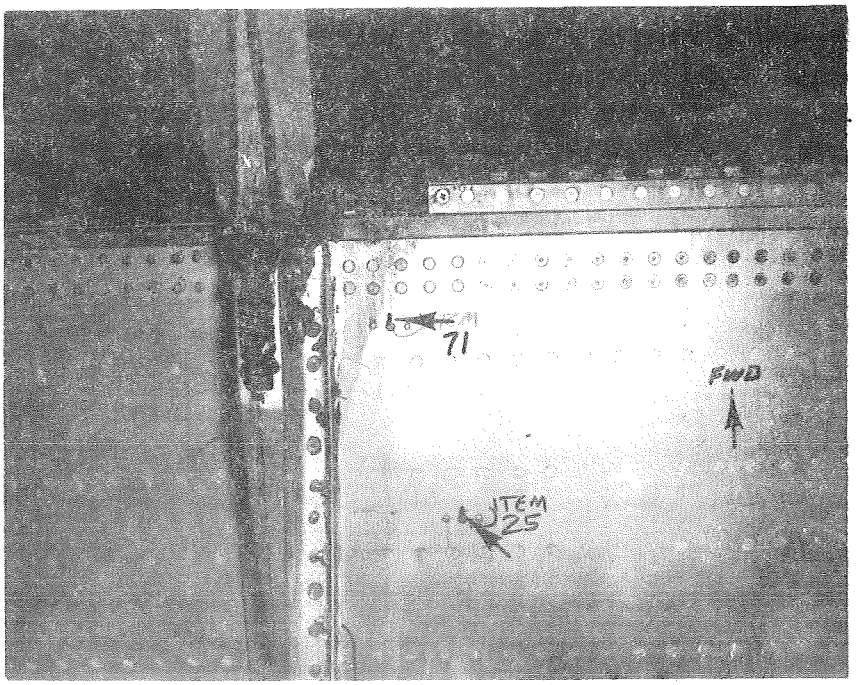


Figure 14-19 Cracks 25 and 71 originated in drain holes at WS 171R and 173R.

APPENDIX A

to

PART G

GENERAL SECTION

SPECIMEN STRUCTURAL DETAILS

This appendix will provide background knowledge on the structural details of the specimens tested, especially in those areas where significant damage has been experienced. Also, infomation is provided on the structural details of the repairs encountered during the conduct of the test program.

Lower Surface

The basic material used in the stringers, skin, and reinforcing structure of the lower surface is 7075-T6 aluminum. The lower surface basic structure consists primarily of three panels of flat sheets, each of varying thickness extending from WS 219R to WS 219L (438 inches long) and stiffened by 13 spanwise extruded 'hat' section stringers, which are attached by rivets except at panel splices which use steel lockbolts. The numbering system for stringers of both surfaces is shown in Figure A-5. Due to the presence of lower surface access doors at WS 120.5L and R, this basic structure is reinforced in those areas with a doubler, which extends from WS 168 to WS 83.0, and reinforcement 'beams' which are under the stringer hat sections of Stringers 15, 16, 20 and 21 and extend from WS 182 to WS 59. These structural details are illustrated in Figures A-1 and A-2.

Upper Surface

The basic material used in the stringers and skin of the upper surface is 7178-T6 aluminum, while most of the reinforcement doublers, channels, angles etc. are of 7075-T6 aluminum. The upper surface skin consists of four panels of extruded stock of constant cross section (except near reinforcement structure) which are essentially flat sheets with integral 'risers' spaced every 3.3 inches. Each such panel extends from WS 219L to WS 219R (438 inches long). This skin is stiffened by eleven hat-shaped spanwise stringers, which are attached to the skin with rivets except at panel splices which use steel lockbolts. There are five cutout areas in the upper surface (doors at WS 192R and L, fuel filler holes at WS 120.5R and L and a door at WS 1.5R) and each of these cutouts are reinforced by doublers (composed of 7075-T6 aluminum) and other structure as is illustrated, along with the basic structure in Figure A-4. Location of rib structure is shown in Figure A-6.

Repairs

Several repairs had been installed during the service life of some of the wing boxes, both as repairs for cracks already in existence and as fatigue preventives. Some of these repairs were not removed and were present during testing. These repairs are enumerated in Table A-1 and are illustrated in Figures A-3 and A-4.

TABLE A-I

REPAIRS PRESENT DURING TESTING
(See Figures A-1 through A-4)

UPBENDING SPECIMENS

Spec. No.	
1	No lower surface repairs
2	No lower surface repairs
3	No lower surface repairs
4	No lower surface repairs
5	Repair #1 present at all outboard ends of door reinforcement beams Repair #6 present at two fasteners at each of the corners of both WS 120.5 fuel bag access doors.
13	Repair #3 present at outboard ends of door reinforcement beams at Stringers 15, 16 and 21 and Stringer 20, LHS only. Repair #2 present at outboard end of RHS Stringer 20 beam. Repair #5 present at inboard ends of door reinforcement beams, covering Stringers 14 thru 18. On the left hand side, all straps and doublers were cut at WS 62L and the inboard portions removed, and fasteners common to stringers, splices, and reinforcement beams replaced. Repair #4 present at inboard ends of door reinforcement beams, covering Stringers 20 and 21. On the left hand side, all straps and doublers were cut at WS 62L and removed as above.
14	Repair #3 present at outboard ends of door reinforcement beams at Stringers 15, 16 and 21, both sides. Repair #2 present at outboard ends of door reinforcement beams at Stringer 20, both sides. Special Configuration: Door attachment fastener holes not countersunk, either side. Repair #4 present at inboard ends of the door reinforcements under Stringers 15 and 16. On the RHS the repair on the right hand side was cut at WS 62R, and the inboard portion removed, as in Specimen #13. Repair #5 present at inboard ends of the door reinforcement beams at Stringers 20 and 21 covering Stringers 19 thru 22. On the Right hand side, this repair was cut at WS 62R, and the inboard portion removed, as in Specimen #13.

TABLE A-1 (Continued)

REPAIRS PRESENT DURING TESTING
(See Figures A-1 through A-4)

DOWNBENDING SPECIMENS

Spec. No.	
6	Repair No. 9 installed on both fuel filler holes.
7	No upper surface repairs
8	Repair No. 7 installed at WS 34.5L and WS 37.5R Repair No. 8 installed at WS 120.5R & L
9	No upper surface repairs
10	Repair No. 7 installed at WS 34.5L and WS 37.5R Repair No. 9 installed at WS 120.5 R and L
11	Repair No. 10 at aft outboard corner of the door cutout at WS 192L was removed prior to testing. Repair No. 10 at fwd inboard, and the aft outboard corners of the door at WS 192R was removed Repair No. 8 installed at WS 120.5 R and L Repair No. 7 installed at WS 34.5L and 37.5R.
12	Repair No. 9 removed from WS 120.5 R & L prior to testing.

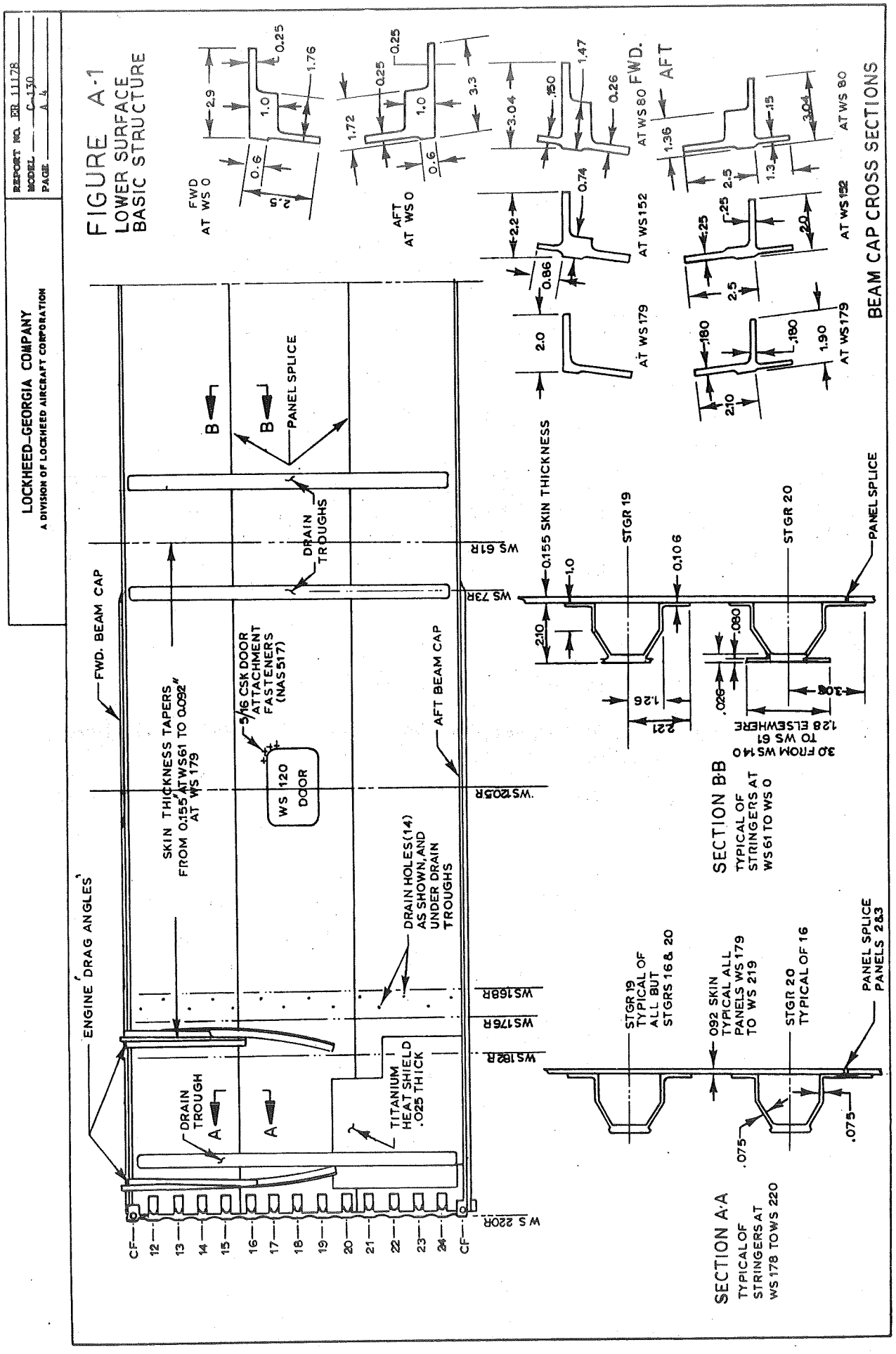
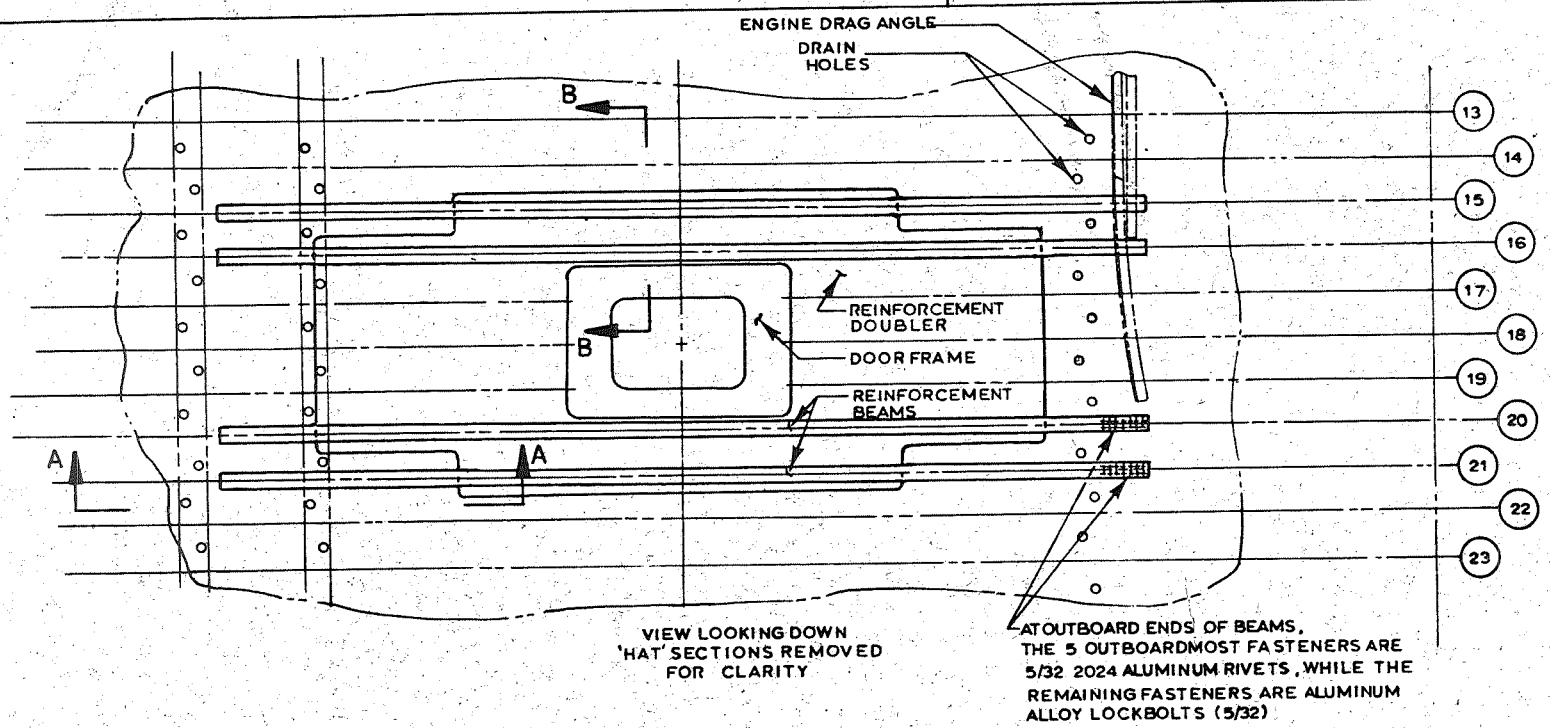
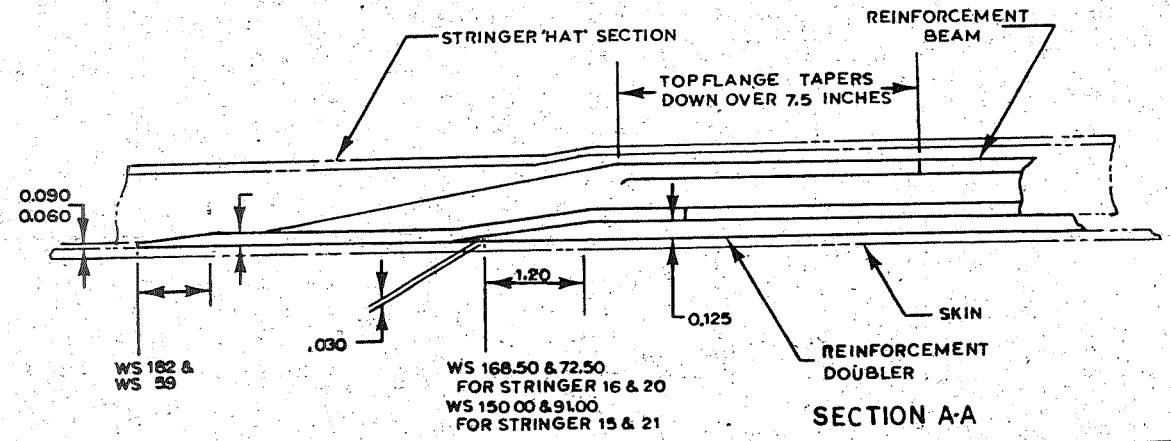
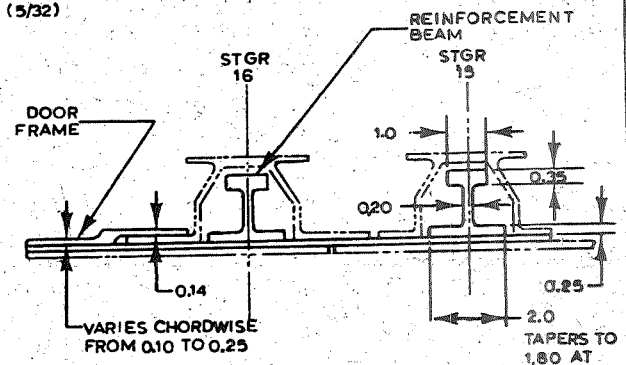

LOCKHEED-GEORGIA COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
REPORT NO. ER 11178
MODEL C-130
PAGE A-4
FIGURE A-1
LOWER SURFACE
BASIC STRUCTURE
FWD. BEAM CAP
AFT BEAM CAP
ENGINE DRAG ANGLES
SKIN THICKNESS TAPERS FROM 0.155 AT WS61 TO 0.092" AT WS 179
5/16 CSK DOOR ATTACHMENT FASTENERS (NAS517)
WS 120 DOOR
DRAIN TROUGHS
PANEL SPLICE
DRAIN HOLES(14) AS SHOWN, AND UNDER DRAIN TROUGHS
TITANIUM HEAT SHIELD .025 THICK
WS 61R
WS 73R
WS 120R
WS 168R
WS 176R
WS 182R
WS 220R
CF 12 13 14 15 16 17 18 19 20 21 22 23 24 CF
SECTION A-A
TYPICAL OF STRINGERS AT WS 178 TO WS 220
STGR 19 TYPICAL OF ALL BUT STGRS 16 & 20
.092 SKIN TYPICAL ALL PANELS WS 179 TO WS 219
STGR 20 TYPICAL OF 16
PANEL SPLICE PANELS 2&3
.075
SECTION B-B
TYPICAL OF STRINGERS AT WS 61 TO WS 0
0.155 SKIN THICKNESS
STGR 19
0.106
STGR 20
PANEL SPLICE
30 FROM WS 140 TO WS 61
128 ELSEWHERE
BEAM CAP CROSS SECTIONS
FWD AT WS 0
AFT AT WS 0
AT WS 179
AT WS 152
AT WS 80
FWD.
AFT
GA FORM 4358-1

FIGURE A-2
WS 120.5 LOWER SURFACE DOOR REINFORCEMENT STRUCTURE



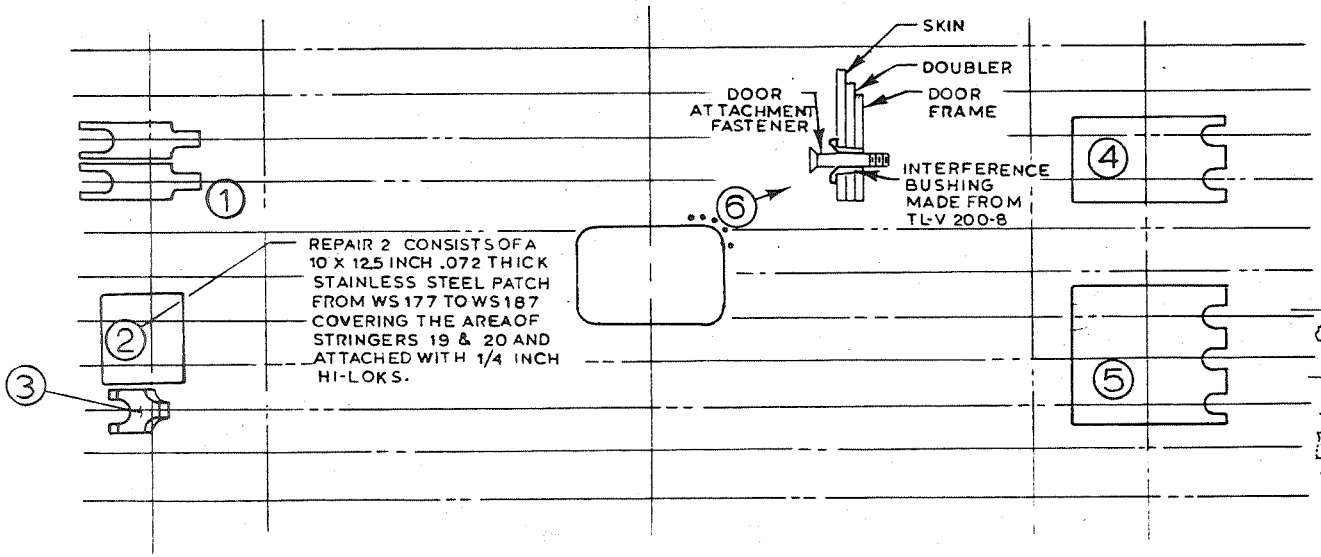


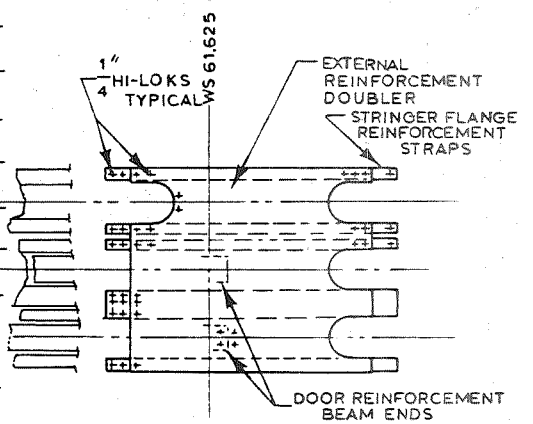
SECTION A-A

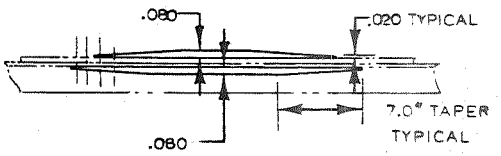


SECTION B-B
(ROTATED 90° CW)

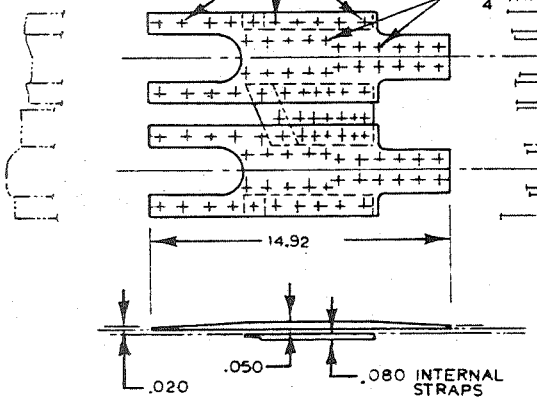
FIGURE A-3
LOWER SURFACE REPAIRS

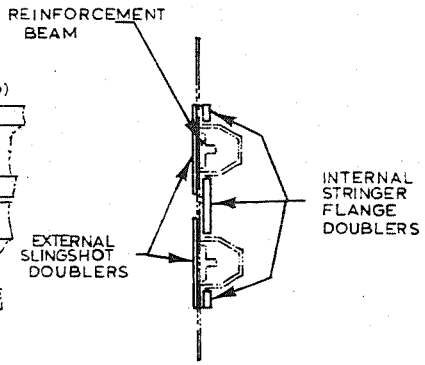




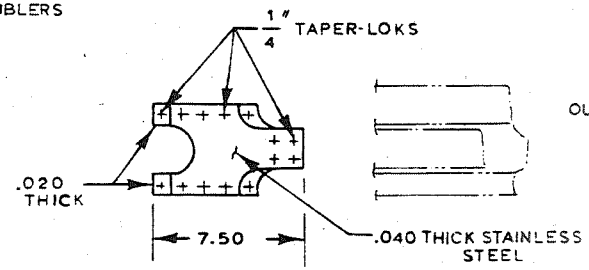


REPAIRS 4 & 5
INBOARD ENDS OF BEAMS





REPAIR 1
OUTBOARD ENDS OF BEAMS
STEEL STRAP AND DOUBLER REPAIRS



REPAIR 3
OUTBOARD ENDS OF BEAMS

FIGURE A-4
UPPER SURFACE STRUCTURE AND REPAIRS

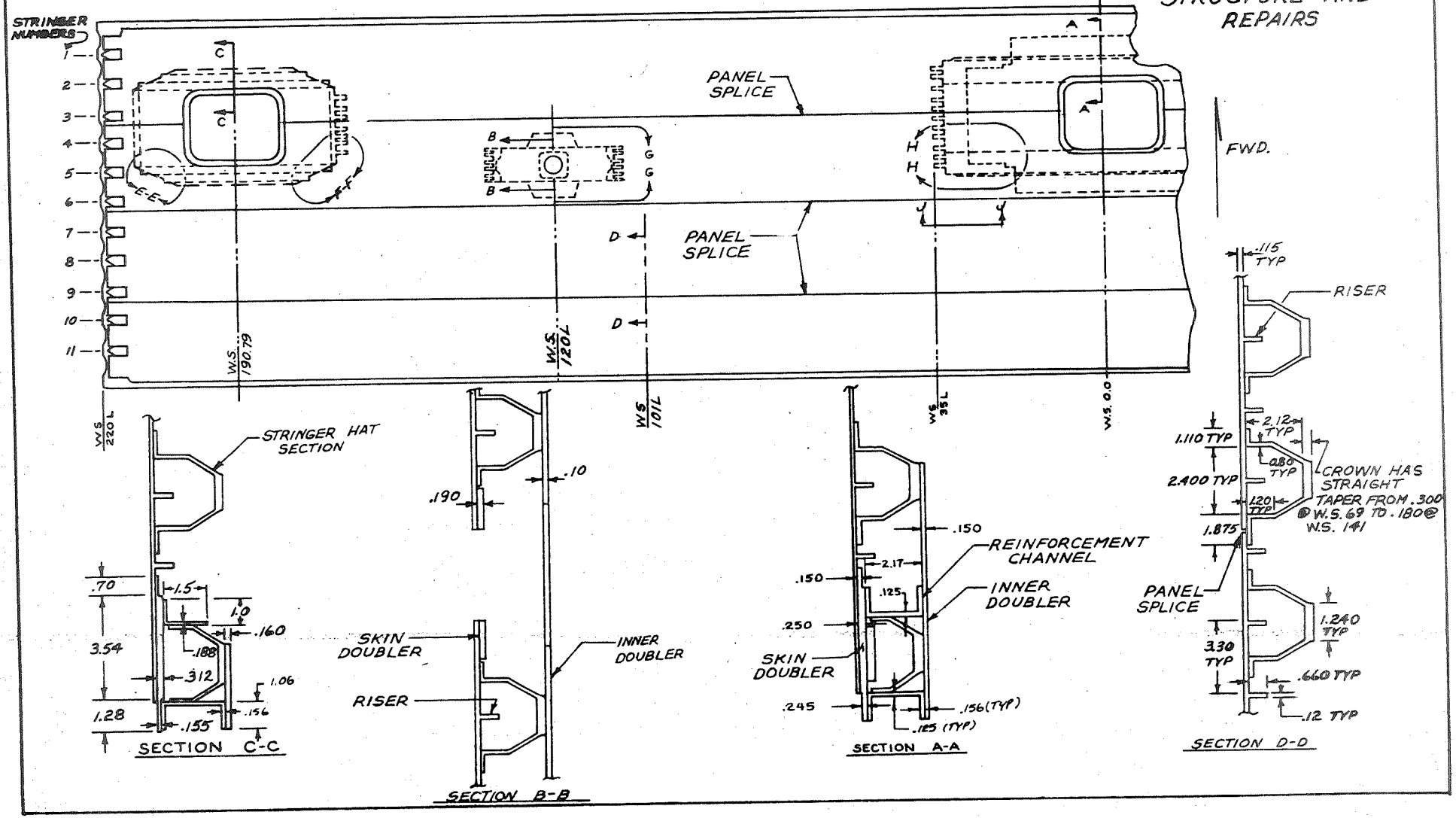
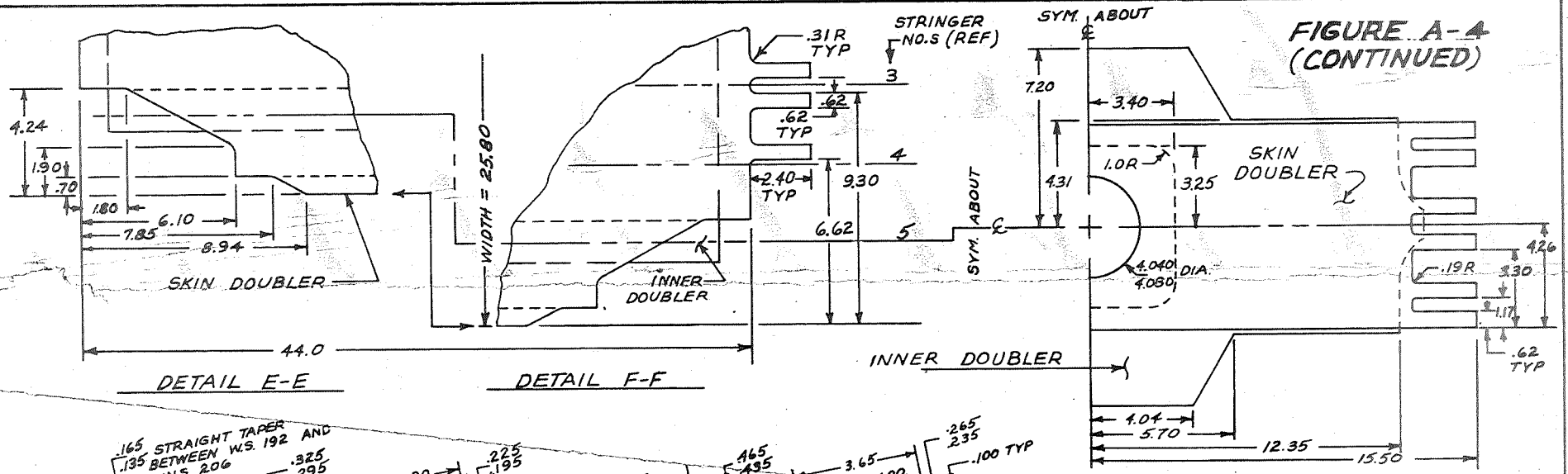
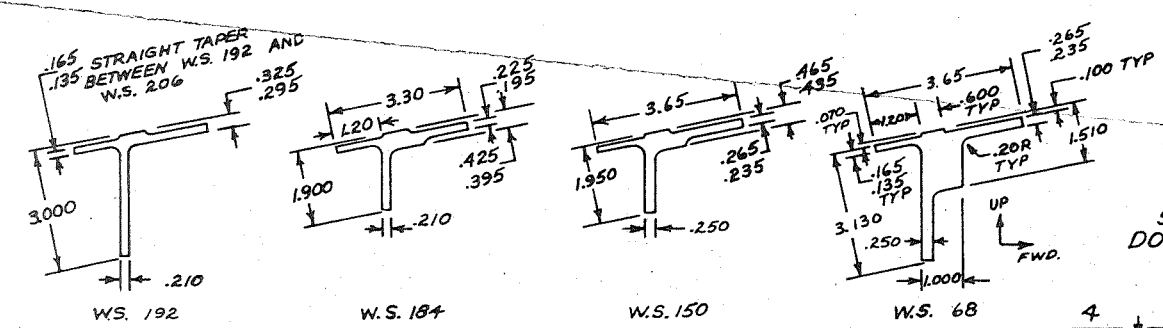
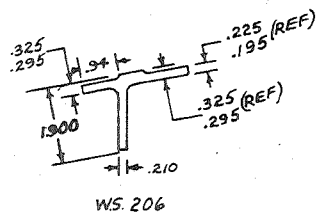


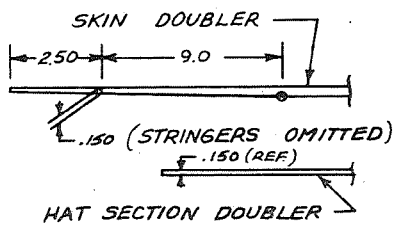
FIGURE A-4 (CONTINUED)





UPPER REAR BEAM CAP CROSS SECTIONS





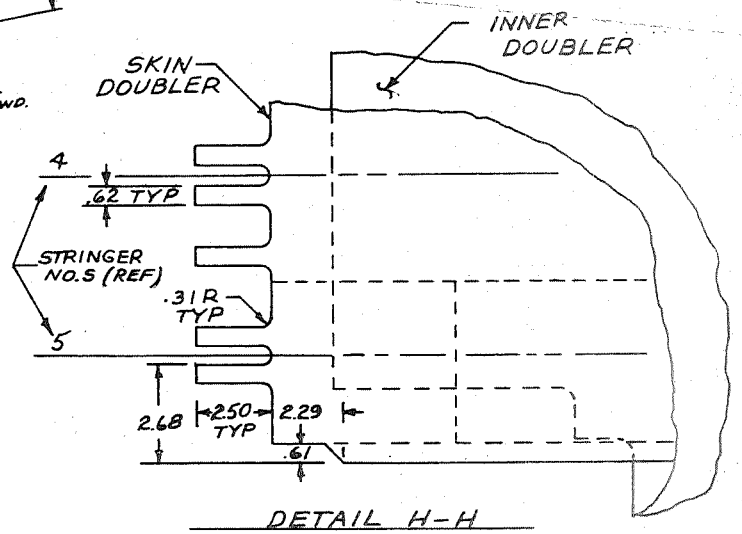
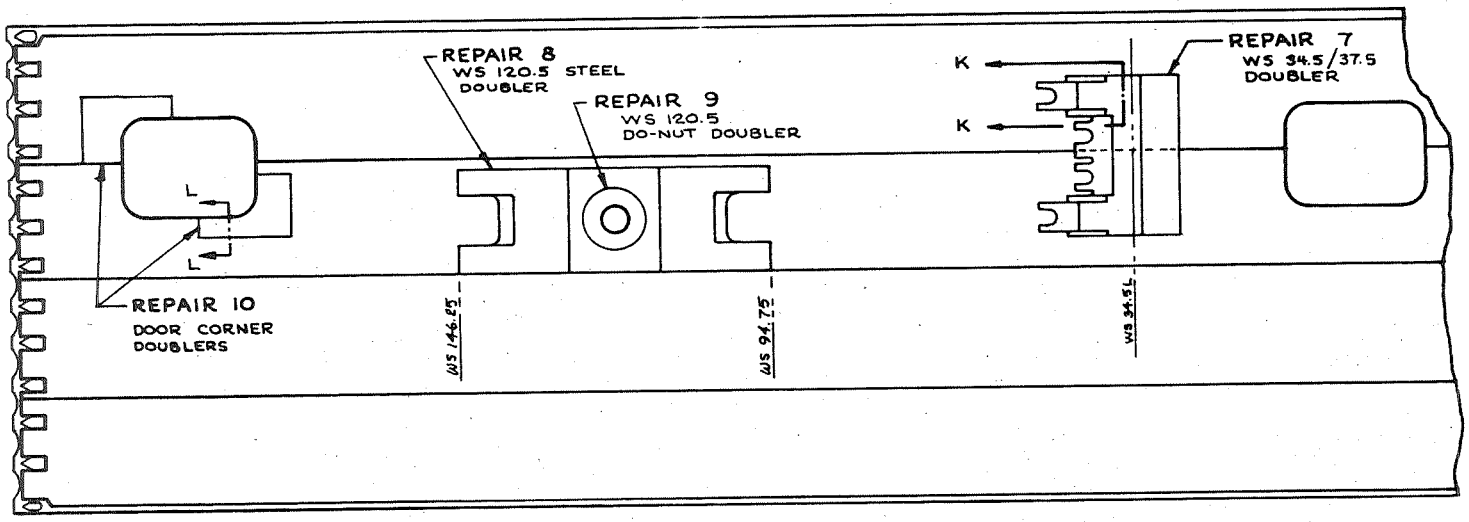
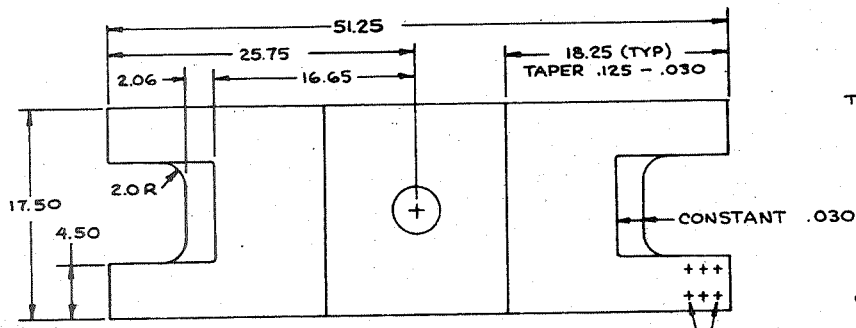
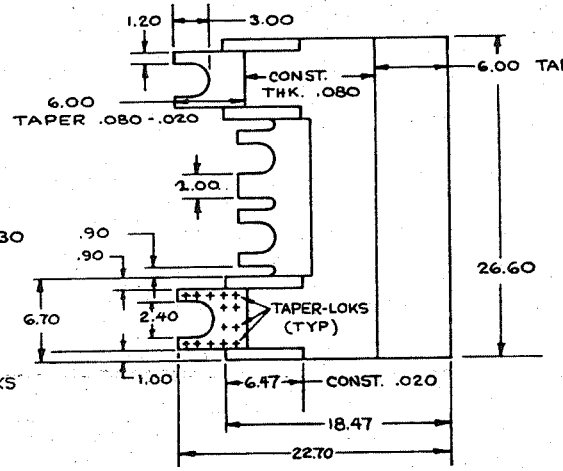


FIGURE A-4
CONTINUED

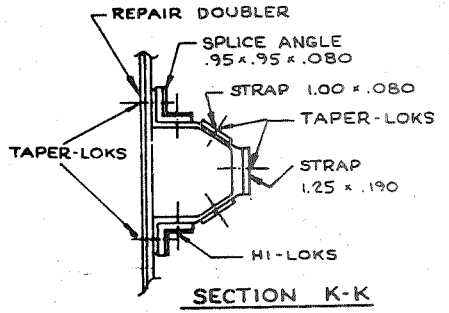




DETAIL WS 120.5 STEEL DOUBLER

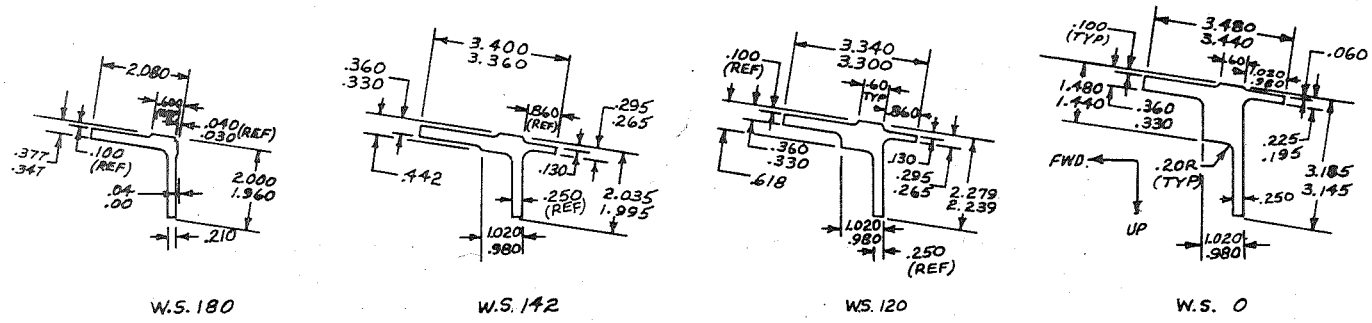


DETAIL WS 34.5/37.5 DOUBLER



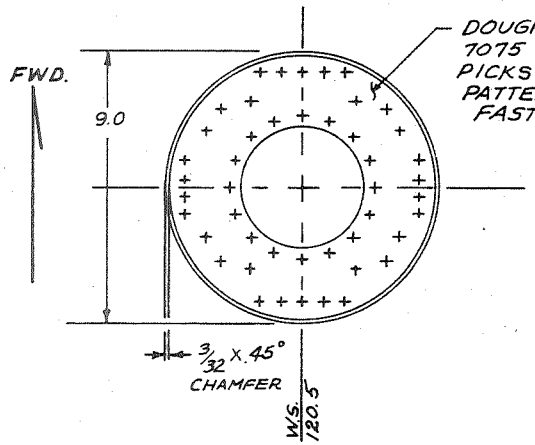
SECTION K-K

FIGURE A-4
CONTINUED

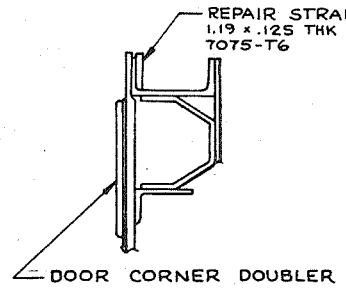


W.S. 180 W.S. 142 W.S. 120 W.S. 0

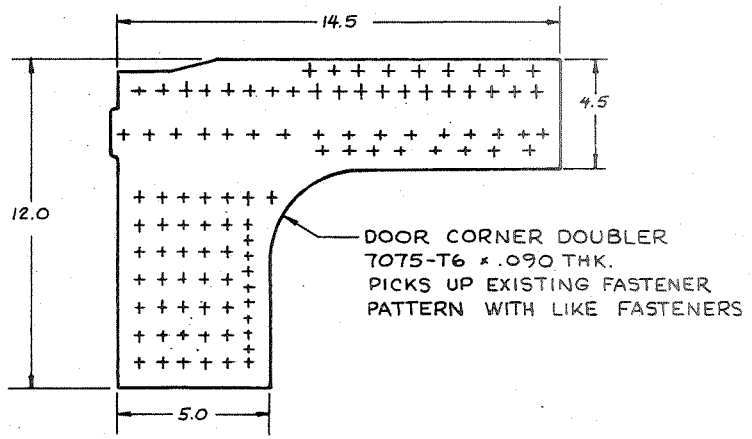
LOWER REAR BEAM CAP CROSS SECTIONS



REPAIR NO. 9



SECTION L-L



REPAIR Nº 10

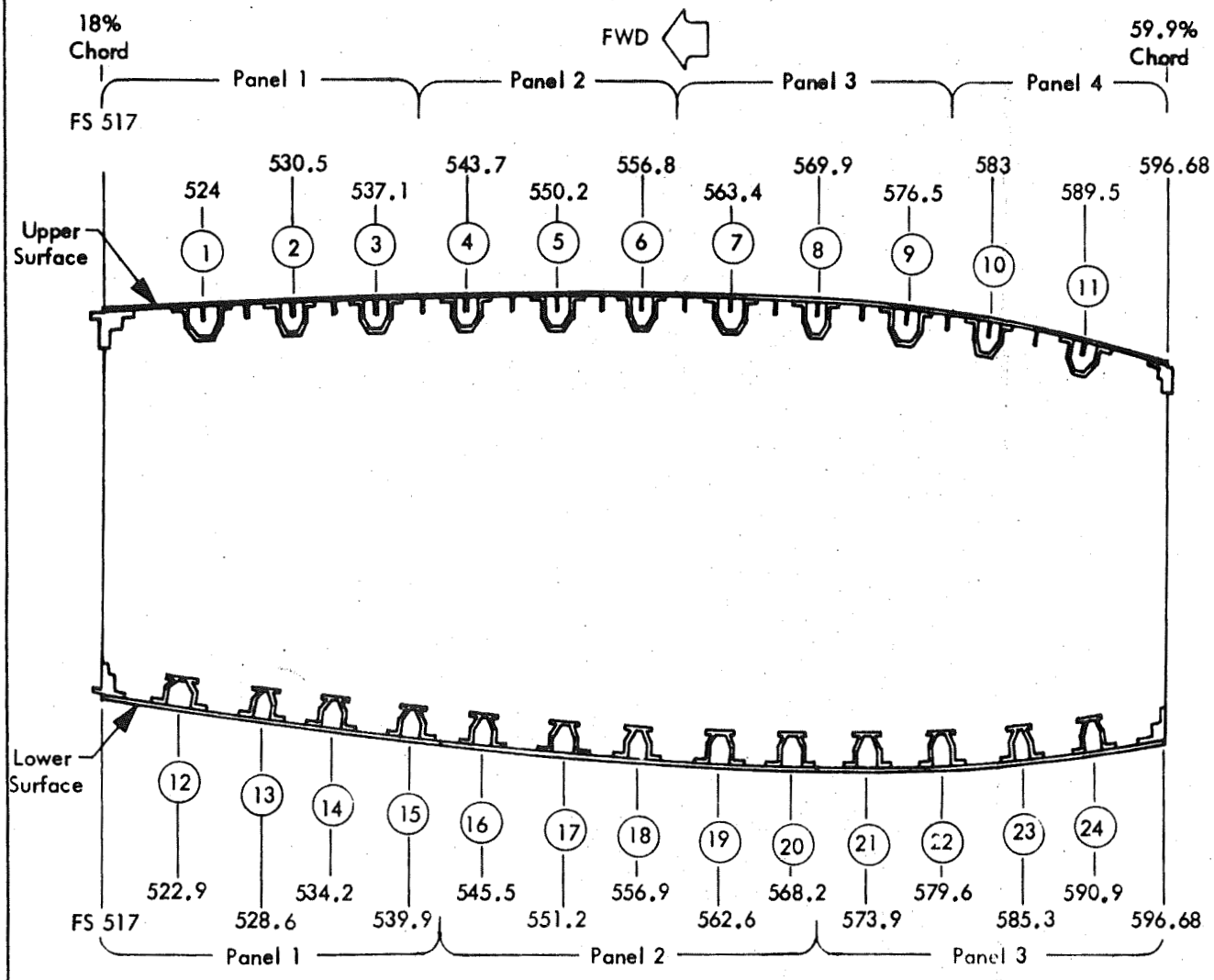


Figure A-5 · General cross-section of center wing showing locations of stringers and skin panels.

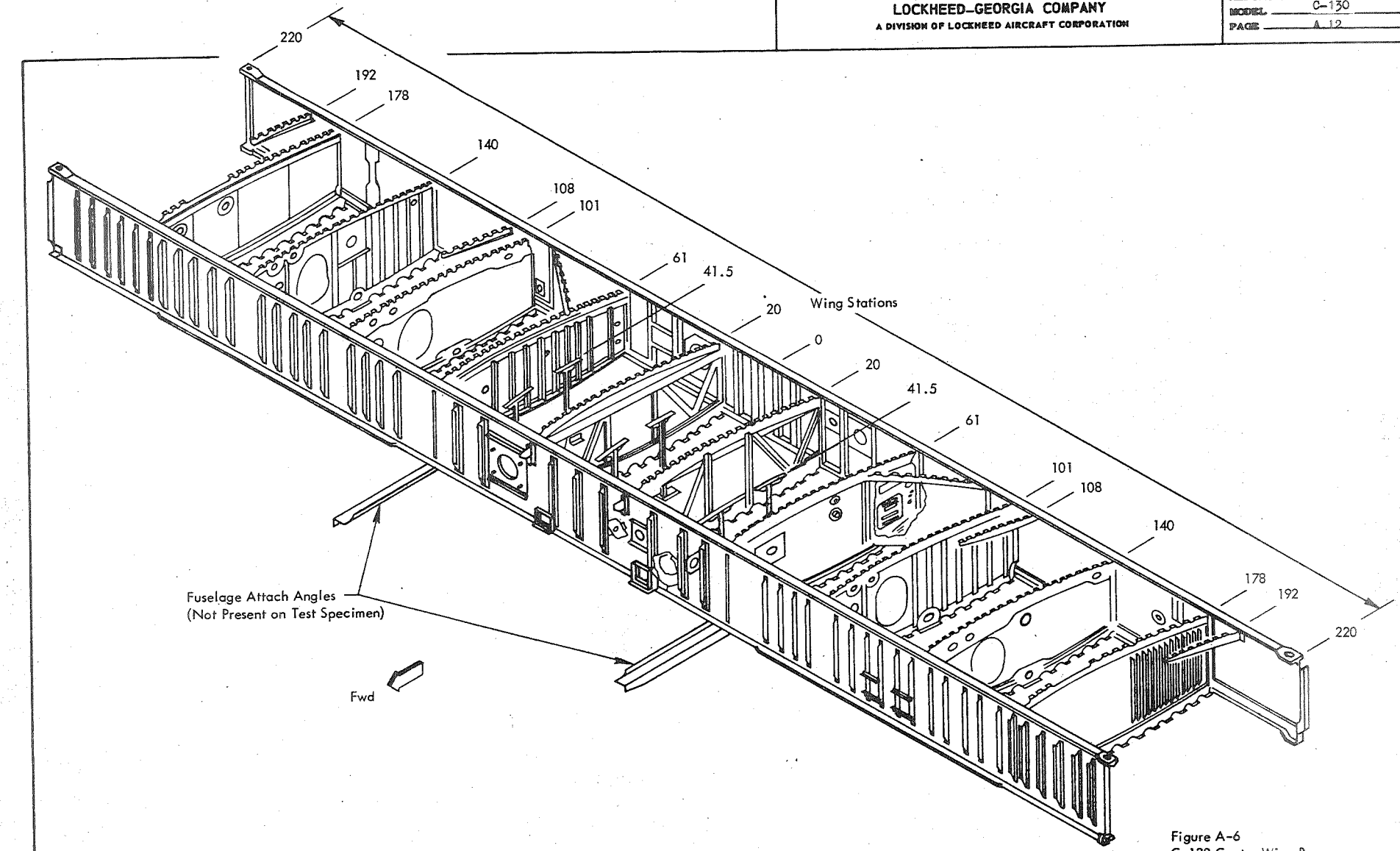


Figure A-6
C-130 Center Wing Box
Beam and Rib Structure

Appendix B

to

Part G

General Section

Flight Service Experience

APPENDIX B FLIGHT SERVICE EXPERIENCE

In order to provide a means of comparison between loads applied in the residual strength and crack propagation tests and the actual loads occasioned by service usage, it is necessary to provide some estimation of aircraft usage and the corresponding service loads. Since the C-130E is a widely used aircraft, and has been the subject of hundreds of thousands of manhours of engineering study, a great deal of information has been accumulated dealing with loads and usage. For the purposes of this report, however, a greatly simplified description of these data should prove sufficient.

In order to describe the usage of the aircraft by the United States Air Force, nine basic mission types are described as follows:

#1 Stewart (or Basic) Pilot Training: Includes touch-and-go landings but no high speed, low altitude continuous flights.

#2 Proficiency Training: Includes both touch-and-go and stop-and-go landings, but no high speed, low altitude continuous flights.

#3 Shuttle: Routine logistics flights of under $1\frac{1}{2}$ hours duration.

#4 Short Range Logistics: Routine logistics flights between $1\frac{1}{2}$ and $4\frac{1}{2}$ hours duration.

#5 Long Range Logistics: Routine logistics flights of greater than $4\frac{1}{2}$ hours duration.

#6 Air Drop: Mission includes aerial delivery.

#7 Storm Reconnaissance: Primary function is performance of weather reconnaissance flights, usually of long duration.

#8 Combat Training (or Support) Miscellaneous Missions, similar to Logistics, including Low Level Maneuvers.

#9 Low Level (Sky Hook): Low level, High speed dash.

Figures B-1 through B-9 show the variation of certain flight parameters (fuel, cargo, airspeed and altitude) with flight time. These figures will serve to provide a better general idea of what usage each of these nine missions involves.

Table B-1 shows the number of flight hours accumulated by each aircraft from which a specimen was taken (except for specimens 7, 13 and 14, which were not tracked) and the number of flight hours accumulated in each mission.

Loads monitoring for the two Delta airlines wing boxes (specimens number 13 and 14) was not as comprehensive as for the military aircraft, but a single type of mission for these commercial aircraft is reported as being predominant. This mission consists of $1\frac{1}{2}$ hour flights with 20,000 lbs of cargo, initial weight of 20,000 to 25,000 lbs of fuel at take off and flight at 17,000 ft altitude. Characteristics of this mission fall roughly between those of military aircraft missions No. 3 and No. 4.

In the comprehensive analysis performed by Lockheed for the Air Force each mission has been broken up into mission segments, such as gust, taxi, landing impact, etc., which separate a flight into distinct types of loading. A typical mission consists of several segments, each associated with a corresponding mean moment distribution on the wing. There could, moreover, be more than one example of each particular type of segment, for instance, there might be several taxi segments, each with different mean moment distribution, reflecting, say, the change in the amount of fuel contained in the wings. In addition to mean moments, each such segment would also be associated with a set of numbers representing the statistical probability of the occurrence of a transient incremental moment of any one of several magnitudes, as calculated from flight test data. The loads history of a given type mission at each particular wing station is then expressed as a series of mission segments each associated with a given mean moment and a set of numbers representing the probabilities of varying transient incremental moment magnitudes. From these data and the stress to moment ratios given in Figure 9, theoretical damage may be statistically calculated using any fatigue theory desired. (Generally Miner's Cumulative Damage Theory is used at Lockheed.)

For the purpose of this report, these mission segments have been consolidated within the missions into seven particular types; taxi, landing impact, landing rollout, take off, maneuver at altitudes greater than 500 feet, maneuver at altitudes less than 500 feet, and gust. The corresponding mean moments given in Table B-II, are the time weighted averages of all mission segments of that type within each mission. The transient incremental moment probabilities are summed for each type segment within each mission, and these probabilities are given in Table B-III. The numbers in these tables, taken together, can be used to give a good approximation of the mission profiles. For instance, in the Stewart Pilot Training Mission (No. 1), the gust segment has a mean load of 4.26 million inch-pounds at W.S. 120.5. Superimposed on this mean load are transient incremental loads whose probability of occurrence (given in Table B-III) decreases with magnitude. So, during this 5 hour mission, all but 7089.4 transient moments due to gust at W.S. 120.5 will be less than 0.5×10^6 in-pounds, of these all but 1206.3 will be less than 1.0×10^6 in-pounds, of these, all but 230.6 will be less than 1.5×10^6 in-pounds, etc. These values are given for all missions at two wing stations, W.S. 120.5 and W.S. 181.

In conjunction with Table A.I these mission profile data may be used to describe theoretical damage incurred during service. Table A.I describes the total cumulative usage of each aircraft in each mission. For instance specimen No. 1 has experienced 66 flight hours of Mission No. 1, which is equivalent to 16.5 missions.

TABLE B-I. C-130E FLIGHT SERVICE RECORD - FLIGHT HOURS & % UTILIZATION PER MISSION

Location (1 Oct 1970)	Aircraft Serial Number	Test Specimen Number	Total Hours	Mission number 1 Hours	1 %	2 Hours	2 %	3 Hours	3 %	4 Hours	4 %	5 Hours	5 %	6 Hours	6 %	7 Hours	7 %	8 Hours	8 %	9 Hours	9 %
Ching Chaun Kang * (Sewart 1000, SEA →)	64-500	1	5920	66	1	217	4	2262	38	2401	41	603	10	265	4	0	0	53	1	53	1
Pope	63-7838	2	4020	299	7	423	11	248	6	1188	30	976	24	394	10	0	0	244	6	248	6
Ching Chaun Kang *(Sewart 900, SEA →)	64-514	3	6094	77	1	216	4	2280	38	2501	41	634	10	258	4	0	0	64	1	64	1
Forbes *(Sewart 3600) *(Mildenhall 300) *(Forbes →)	64-539	4	4493	389	9	397	9	334	7	1075	24	1267	28	317	7	0	0	352	8	362	8
Ching Chaun Kang * (Sewart 1000, SEA →)	64-534	5	6296	99	2	170	3	2282	36	2236	36	1149	18	196	3	0	0	82	1	82	1
Langley * (Travis 4400) * (Pope 600) * (Langley →)	62-1828	6	7374	204	3	646	9	215	3	999	14	4496	61	436	6	0	0	152	2	226	3
	58-5397	7	5617	Data unavailable - early airplanes not tracked																	
Ching Chaun Kang * (Eglin 800) * (Dyess 750) * (SEA →)	62-1859	8	7892	72	1	217	3	2712	34	3371	43	1268	16	232	3	0	0	5	-	15	-
Ching Chaun Kang * (Charleston 5000) * (SEA →)	62-1795	9	11251	0	0	830	7	1774	16	1377	12	6929	62	341	3	0	0	0	0	0	0
Ramstein * (Charleston 4000) * (Pope 1600) (SEA 200) * (Ramstein →)	64-572	10	6603	225	3	444	7	247	4	147	2	4490	68	258	5	0	0	429	6	363	5
Ching Chaun Kang * (Pope 900) * (SEA →)	63-9811	11	7660	94	1	247	3	2940	38	2865	37	1134	15	276	4	0	0	52	1	52	1
Pope * (Eglin 900) * (Sewart 1150) * (SEA 1000) * (Pope →)	62-1860	12	5674	380	7	1130	20	725	13	1895	33	1089	19	185	3	0	0	125	2	145	3
Delta Air Lines	N9268R	13	13000 (Approx)	Not tracked, but principal utilization was marginal between Missions 3 and 4.																	
Delta Air Lines	N9259R	14	13000 (Approx)	Not tracked, but principal utilization was marginal between Missions 3 and 4.																	

* Previous locations and flight hours

C-130 E = 72,892 Lb. Equipped Weight Empty
155,000 Lb. Maximum Normal Gross Weight at Takeoff
175,000 Lb. Maximum Emergency Gross Weight at Takeoff

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TABLE B-II

MISSION LOAD PROFILES

MEAN LOADS (a)

Mission Name	Wing Sta.	Mission Segment Mean Moments, In.-Lb x 10^{-6} Gust	Man.(b)	Man.(b)	Take-Off	Land.Roll	Land.Imp.	Taxi
1 Stewart (Basic) Pilot Train.	120.5	4.26		4.68	-4.60	-5.55	3.13	-7.01
	181	3.72	N/A	4.10	-3.38	-4.15	2.45	-5.33
2 Proficiency Train.	120.5	4.75		4.93	-4.89	-5.71	3.43	-7.13
	181.	4.12	N/A	4.28	-3.62	-4.29	2.69	-5.44
3 Shuttle	120.5	6.20		6.76	-5.86	-6.12	3.89	-7.35
	181	5.25	N/A	5.77	-4.41	-4.62	3.05	-5.62
4 Short Range Logistic	120.5	6.78		7.45	-5.99	-4.72	4.37	-6.73
	181	5.73	N/A	6.26	-4.52	-3.49	3.44	-5.12
5 Long Range Logistic	120.5	6.80		7.48	-7.10	-3.82	4.60	-6.83
	181	5.77	N/A	6.34	-5.28	-2.78	3.63	-5.14
6 Air Drop	120.5	4.87		5.24	-5.85	-6.30	3.25	-7.60
	181	4.19	N/A	4.51	-4.40	-4.76	2.53	-5.82
7 Storm Reconnaissance	120.5	5.60		6.15	-8.15	-3.81	3.71	-7.35
	181	4.82	N/A	5.29	-5.98	-2.77	2.93	-5.49
8 Combat Train.	120.5	5.07	5.65	5.64	-5.46	-5.64	3.33	-7.36
	181	4.35	4.79	4.70	-4.07	-4.37	2.61	-5.61
9 Sky Hook (Low Level)	120.5	5.23	5.58	5.67	-6.58	-6.63	3.27	-7.97
	181	4.48	4.71	4.86	-4.95	-5.03	2.54	-6.10

(a) All mean loads within mission segments are calculated and are time weighted averages.

(b) Maneuver at altitudes less than 500 feet.

(c) Maneuver at altitudes greater than 500 feet.

TABLE B-III

MISSION LOAD PROFILES

TRANSIENT LOAD INCIDENCES (a)

MISSION	Moment Inch Pounds X 10^{-6}	Incidences Per Mission, ΣN : Due to Mission Segments						
		Gust	Man.(b)	Man.(c)	Take-Off	Land. Roll	Land. Imp.	Taxi
1	@ W.S. 120.5							
	0.5	7,089.4	0.0	594.2	597.5	414.9	33.0	1,487.1
	1.0	1,206.3		94.1	147.7	130.2	31.2	117.3
Stewart	1.5	230.6		17.8	37.1	42.4	26.3	14.2
(Basic)	2.0	49.2		4.1	9.5	14.1	19.9	2.3
Pilot	2.5	11.7		1.1	2.5	4.8	13.4	0.4
Training	3.0	3.1		0.4	0.7	1.7	8.1	0.1
	3.5	1.0		0.1	0.2	0.6	4.4	0
Duration:	4.0	0.3		0.1	0.1	0.2	2.1	0
	4.5	0.1		0.0	0	0.1	1.0	0
4 hrs.	5.0	0.1		0.0	0	0	0.4	0
			0.0					
	@ W.S. 181							
	0.5	7,157.8	0.0	594.2	603.5	416.9	33.0	1,494.6
	1.0	921.3		75.8	110.0	101.4	30.5	71.7
	1.5	138.2		11.9	20.6	26.0	24.2	6.3
	2.0	24.0		2.4	4.0	6.9	16.4	0.7
	2.5	4.8		0.6	0.8	1.8	9.6	0.1
	3.0	1.2		0.2	0.2	0.5	4.8	0
	3.5	0.3		0.1	0	0.2	2.1	0
	4.0	0.1		0	0	0	0.8	0
	4.5	0		0	0	0	0.3	0
	5.0	0	0.0	0	0	0	0.1	0
2	@ W.S. 120.5							
	0.5	5,814.6	0.0	501.2	630.5	562.5	30.0	3,596.5
	1.0	1,102.8		93.2	164.2	180.7	28.7	319.7
	1.5	241.4		21.7	44.4	61.5	24.0	39.8
	2.0	60.2		6.2	12.6	22.1	18.2	6.4
Proficiency	2.5	17.0		2.1	3.8	8.4	12.4	1.2
Training	3.0	5.4		0.8	1.2	3.3	7.6	0.3
	3.5	1.9		0.3	0.4	1.4	4.2	0.1
	4.0	0.7		0.1	0.1	0.6	2.1	0
	4.5	0.3		0.1	0	0.3	0.9	0
	5.0	0.1		0.0	0	0.1	0.4	0
	@ W.S. 181							
	0.5	5,879.1	0.0	501.2	637.5	566.4	30.0	3,626.3
Duration:	1.0	854.8		75.6	124.4	142.5	27.8	199.7
	1.5	149.7		14.8	25.7	39.0	22.0	17.9
3.6 hrs.	2.0	31.0		3.7	5.7	11.5	15.1	2.2
	2.5	7.6		1.1	1.4	3.7	8.9	0.3
	3.0	2.1		0.4	0.4	1.3	4.5	0.1
	3.5	0.7		0.1	0.1	0.5	2.0	0
	4.0	0.3		0.1	0	0.2	0.8	0
	4.5	0.1		0	0	0.1	0.3	0
	5.0	0	0.0	0	0	0	0.1	0
3	@ W.S. 120.5							
	0.5	691.7	0.0	79.6	81.0	82.1	3.0	1,421.8
	1.0	195.9		19.8	22.7	27.4	2.9	153.3
	1.5	59.7		5.6	6.6	9.6	2.4	21.9
	2.0	19.4		1.8	2.0	3.5	1.9	4.0
Shuttle	2.5	6.4		0.6	0.7	1.3	1.3	0.9
	3.0	2.4		0.2	0.2	0.5	0.8	0.2
	3.5	0.9		0.1	0.1	0.2	0.5	0.1
	4.0	0.4		0	0	0.1	0.3	0
	4.5	0.2		0	0	0	0.1	0
	5.0	0.1	0.0	0	0	0	0.1	0
	@ W.S. 181							
	0.5	698.0	0.0	79.6	82.0	83.2	3.0	1,439.4
Duration:	1.0	160.7		16.1	17.6	22.1	2.8	100.8
	1.5	40.6		3.8	4.0	6.3	2.3	10.6
0.8 hrs.	2.0	11.1		1.0	1.0	1.9	1.6	1.5
	2.5	3.3		0.3	0.3	0.6	1.0	0.3
	3.0	1.1		0.1	0.1	0.2	0.5	0.1
	3.5	0.4		0	0	0.1	0.2	0
	4.0	0.1		0	0	0	0.1	0
	4.5	0.1		0	0	0	0	0
	5.0	0	0.0	0	0	0	0	0

(a) Values calculated from flight test data represent the sum of all such incidences within each type of mission segment.

(b) Maneuver at altitudes less than 500 feet.

(c) Maneuver at altitudes greater than 500 feet.

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TABLE B-III (Continued)

MISSION LOAD PROFILES

TRANSIENT LOAD INCIDENCES (a)

MISSION	Moment Inch Pounds X 10^{-6}	Incidences Per Mission, ΣN ; Due to Mission Segments						
		Gust	Man.(b)	Man.(c)	Take-Off	Land. Roll	Land.Imp.	Taxi
4	@ W.S. 120.5							
	0.5	632.6	0.0	126.0	81.3	94.2	3.0	1,522.4
	1.0	184.1		25.0	21.9	24.6	2.8	127.5
	1.5	58.3		6.1	6.0	6.7	2.2	15.1
	2.0	20.0		1.8	1.7	1.9	1.5	2.4
Short	2.5	7.4		0.6	0.5	0.6	0.9	0.4
Range	3.0	2.9		0.2	0.1	0.2	0.4	0.1
Logistic	3.5	1.2		0.1	0	0.1	0.2	0
	4.0	0.6		0	0	0	0	0
	4.5	0.3		0	0	0	0	0
	5.0	0.1	0.0	0	0	0	0	0
	@ W.S. 181							
	0.5	640.5	0.0	126.0	82.3	94.0	3.0	1,529.5
	1.0	150.8		19.4	16.9	17.9	8.7	77.7
	1.5	39.7		3.9	3.6	3.7	1.9	6.6
Duration:	2.0	11.6		1.0	0.8	0.8	1.1	0.8
2.7 hrs.	2.5	3.8		0.3	0.2	0.2	0.5	0.1
	3.0	1.3		0.1	0	0	0.2	0
	3.5	0.5		0	0	0	0.1	0
	4.0	0.2		0	0	0	0	0
	4.5	0.1		0	0	0	0	0
	5.0	0	0.0	0	0	0	0	0
5	@ W.S. 120.5							
	0.5	1,232.3	0.0	305.4	78.1	94.3	3.0	1,494.0
	1.0	371.1		59.8	26.0	24.4	2.7	153.5
	1.5	118.2		14.1	8.7	6.6	2.2	22.5
	2.0	39.9		4.0	2.9	1.8	1.5	4.2
Long	2.5	14.4		1.4	1.0	0.5	0.9	0.9
Range	3.0	5.5		0.5	0.4	0.2	0.5	0.2
Logistic	3.5	2.3		0.2	0.1	0.1	0.2	0.1
	4.0	1.0		0.1	0	0	0.1	0
	4.5	0.5		0.1	0	0	0	0
	5.0	0.3	0.0	0	0	0	0	0
	@ W.S. 181							
	0.5	1,247.6	0.0	305.3	79.2	94.1	3.0	1,501.5
	1.0	308.1		46.5	21.2	17.8	2.6	100.3
Duration:	1.5	81.6		9.1	5.7	3.6	1.9	11.1
	2.0	23.5		2.3	1.5	0.8	1.1	1.7
7 hrs.	2.5	7.4		0.7	0.4	0.2	0.5	0.3
	3.0	2.5		0.3	0.1	0	0.2	0.1
	3.5	1.0		0.1	0	0	0.1	0
	4.0	0.4		0	0	0	0	0
	4.5	0.2		0	0	0	0	0
	5.0	0.1	0.0	0	0	0	0	0
6	@ W.S. 120.5							
	0.5	3,407.2	0.0	298.4	133.1	254.9	9.0	2,081.9
	1.0	795.0		66.1	33.2	81.2	8.5	204.5
	1.5	210.2		16.8	8.5	26.8	7.3	26.8
	2.0	60.9		4.8	2.2	9.2	5.6	4.5
	2.5	18.9		1.5	0.6	3.2	3.8	0.9
	3.0	6.3		0.5	0.2	1.2	2.4	0.2
Air Drop	3.5	2.2		0.2	0.1	0.4	1.3	0.1
	4.0	0.8		0.1	0	0.2	0.7	0
	4.5	0.3		0	0	0.1	0.3	0
	5.0	0.2	0.0	0	0	0	0.1	0
Duration:	@ W.S. 181							
	0.5	3,440.6	0.0	298.4	134.3	257.9	9.0	2,109.3
1.9 hrs.	1.0	640.4		53.9	24.8	64.6	8.4	131.2
	1.5	140.2		11.4	4.7	17.0	6.7	12.5
	1.0	34.3		2.8	0.9	4.7	4.6	1.6
	2.5	9.2		0.8	0.2	1.4	2.8	0.3
	3.0	2.7		0.3	0	0.4	1.4	0
	3.5	0.9		0.1	0	0.1	0.6	0
	4.0	0.3		0	0	0	0.3	0
	4.5	0.1		0	0	0	0.1	0
	5.0	0.1	0.0	0	0	0	0	0

(a) Values calculated from flight test data represent the sum of all such incidences within each type of mission segment.

(b) Maneuver at altitudes less than 500 feet.

(c) Maneuver at altitudes greater than 500 feet.

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TABLE B-III (Continued)

MISSION LOAD PROFILES

TRANSIENT LOAD INCIDENCES (a)

MISSION	Moment Inch Pounds X 10^{-6}	Incidences Per Mission, ΣN ; Due to Mission Segments						
		Gust	Man. (b)	Man. (c)	Take-Off	Land. Roll	Land. Imp.	Taxi
7	@ W.S. 120.5							
	0.5	8,343.0	0.0	968.6	77.9	95.9	3.0	1,505.8
	1.0	2,081.5		196.0	26.9	20.9	2.7	145.0
	1.5	549.1		42.6	9.4	4.8	2.2	22.2
	2.0	151.9		10.2	3.3	1.1	1.4	4.4
Storm	2.5	44.2		2.7	1.2	0.3	0.8	1.0
Reconaissance	3.0	13.6		0.8	0.4	0.1	0.4	0.3
	3.5	4.5		0.3	0.2	0	0.2	0.1
	4.0	1.7		0.1	0.1	0	0.1	0
	4.5	0.7		0	0	0	0	0
	5.0	0.3	0.0	0	0	0	0	0
	@ W.S. 181							
	0.5	8,375.8	0.0	968.6	76.2	95.1	3.0	1,484.4
	1.0	1,705.0		160.3	20.8	14.4	2.7	89.6
Duration:	1.5	373.4		29.2	5.8	2.3	1.9	10.2
10 hrs.	2.0	87.3		6.1	1.6	0.4	1.0	1.6
	2.5	21.8		1.5	0.5	0.1	0.5	0.3
	3.0	6.0		0.4	0.1	0	0.2	0.1
	3.5	1.9		0.1	0	0	0	0
	4.0	0.7		0.1	0	0	0	0
	4.5	0.3		0	0	0	0	0
	5.0	0.1	0.0	0	0	0	0	0
8	@ W.S. 120.5							
	0.5	9,347.3	199.8	567.9	447.3	387.9	21.0	2,902.2
	1.0	2,922.0	69.9	143.8	114.4	116.5	19.8	250.8
	1.5	1,033.3	25.6	43.1	29.8	36.6	16.7	30.7
Combat	2.0	388.0	9.9	14.5	7.9	11.9	12.6	4.9
Training	2.5	151.4	4.0	5.3	2.2	3.9	8.5	0.9
(Support)	3.0	60.9	1.7	2.1	0.6	1.3	5.2	0.2
	3.5	25.4	0.8	0.9	0.2	0.5	2.8	0
	4.0	10.9	0.4	0.4	0.1	0.1	1.4	0
	4.5	4.9	0.2	0.2	0	0.1	0.6	0
	5.0	2.3	0.1	0.1	0	0	0.3	0
	@ W.S. 181							
	0.5	9,426.9	199.8	567.9	452.6	390.2	21.0	2,925.2
	1.0	2,475.6	60.3	119.9	86.5	90.2	19.4	155.9
Duration:	1.5	755.7	19.2	30.7	17.1	22.3	15.3	13.8
3.6 hrs.	2.0	246.8	6.6	8.9	3.5	5.8	10.4	1.7
	2.5	84.3	2.4	2.9	0.7	1.6	6.1	0.3
	3.0	30.0	0.9	1.1	0.2	0.4	3.1	0
	3.5	11.2	0.4	0.4	0	0.1	1.4	0
	4.0	4.4	0.2	0.2	0	0	0.5	0
	4.5	1.8	0.1	0.1	0	0	0.2	0
	5.0	0.8	0	0	0	0	0.1	0
9	@ W.S. 120.5							
	0.5	4,918.3	122.1	298.3	76.1	82.2	3.0	1,374.0
	1.0	1,858.1	47.8	83.2	22.5	25.1	2.8	144.1
	1.5	735.4	19.3	25.1	6.8	7.9	2.4	19.7
	2.0	303.9	8.1	8.2	2.1	2.6	1.8	3.4
	2.5	130.9	3.5	2.9	0.6	0.8	1.3	0.7
	3.0	58.6	1.6	1.1	0.2	0.3	0.8	0.2
Low Level	3.5	27.1	0.7	0.5	0.1	0.1	0.4	0
(Sky Hook)	4.0	13.0	0.4	0.2	0	0	0.2	0
	4.5	6.4	0.2	0.1	0	0	0.1	0
	5.0	3.3	0.1	0	0	0	0	0
	@ W.S. 181							
	0.5	4,939.2	122.1	298.3	77.8	82.9	3.0	1,395.6
Duration:	1.0	1,612.9	41.7	69.5	17.9	19.7	2.8	94.1
2.1 hrs.	1.5	560.0	14.8	17.8	4.2	4.9	2.2	9.3
	2.0	205.9	5.5	5.1	1.0	1.3	1.6	1.3
	2.5	79.8	2.1	1.6	0.3	0.3	0.9	0.2
	3.0	32.4	0.8	0.5	0.1	0.1	0.5	0
	3.5	13.8	0.4	0.2	0	0	0.2	0
	4.0	6.1	0.2	0.1	0	0	0.1	0
	4.5	2.8	0.1	0	0	0	0	0
	5.0	1.3	0	0	0	0	0	0

(a) Values calculated from flight test data represent the sum of all such incidences within each type of mission segment.

(b) Maneuver at altitudes less than 500 feet.

(c) Maneuver at altitudes greater than 500 feet.

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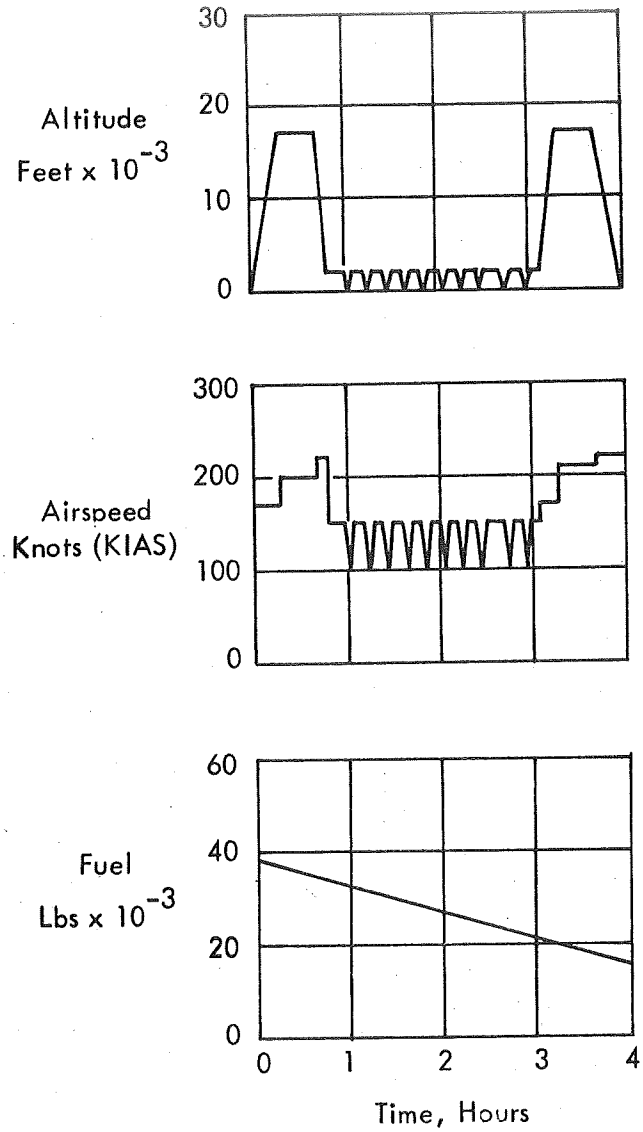
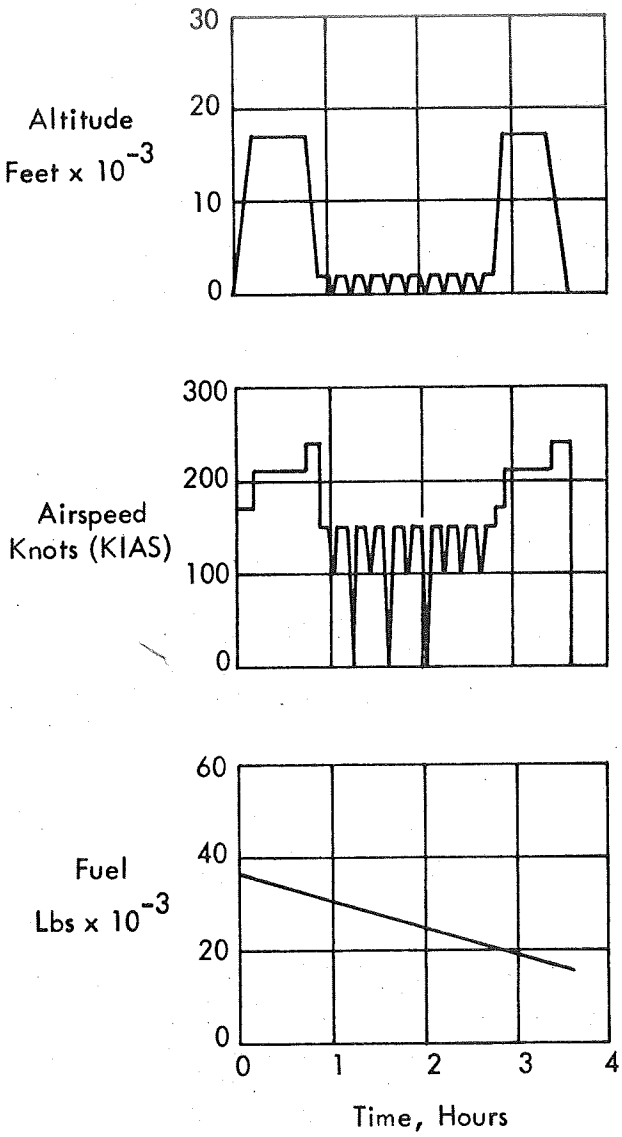


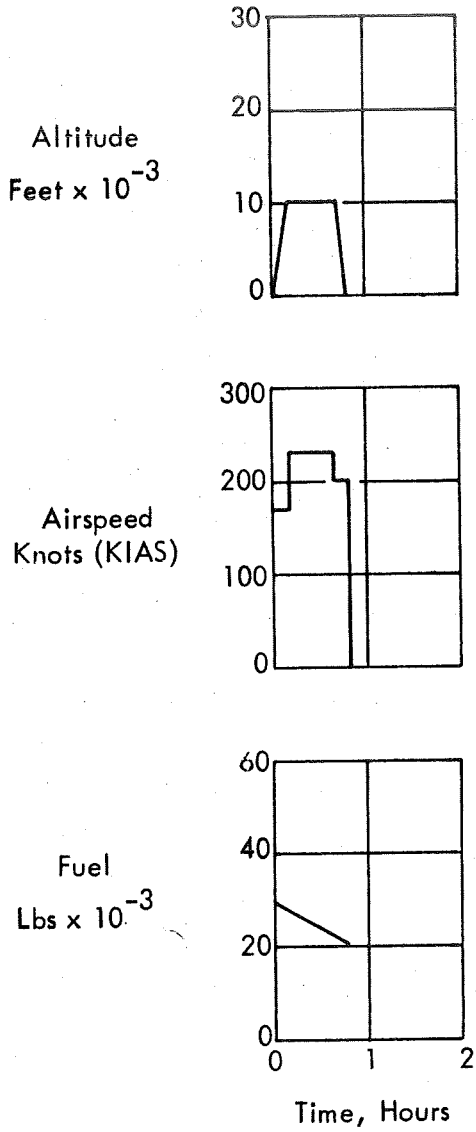
CARGO	1000 LBS
30 - 40	1%
20 - 30	2%
10 - 20	6%
0 - 10	91%

Figure B-1. - Mission 1
Basic (Stewart) Pilot Training Mission



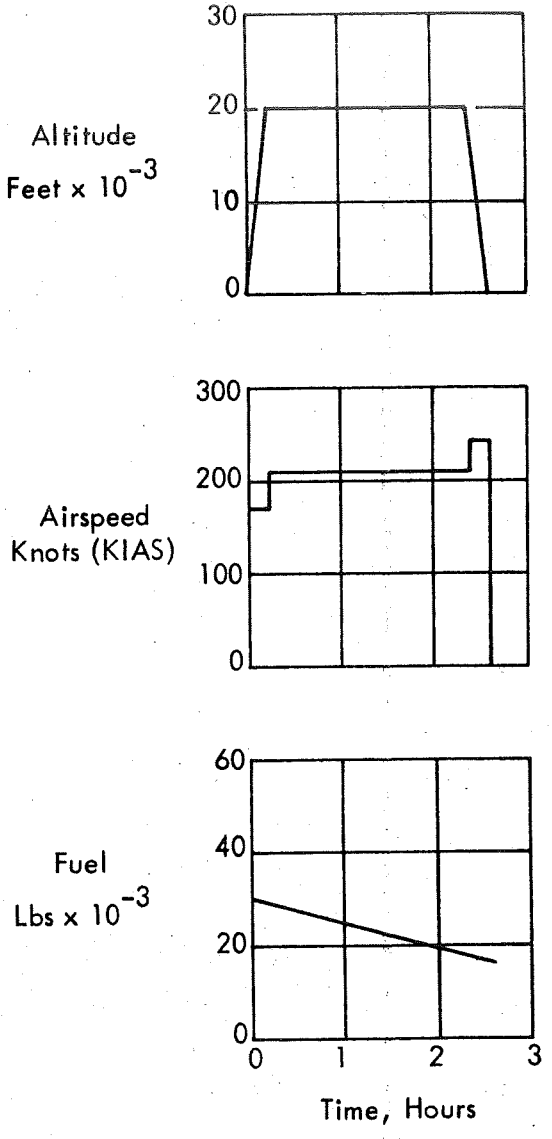
CARGO	1000 LBS
30 - 40	6%
20 - 30	6%
10 - 20	12%
0 - 10	76%

Figure B-2. - Mission 2
Proficiency Training Mission



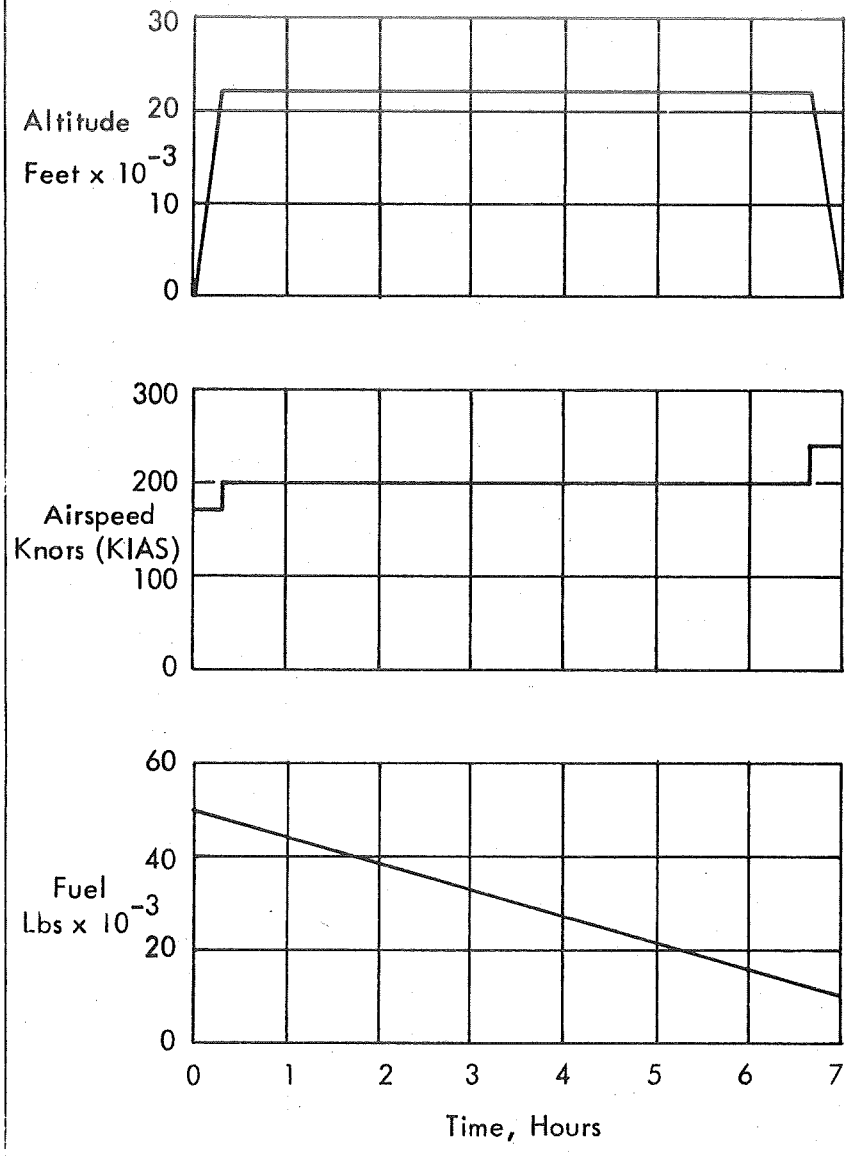
CARGO	1000 LBS
> 40	1%
30 - 40	10%
20 - 30	22%
10 - 20	30%
0 - 10	37%

Figure B-3. - Mission 3 - Shuttle Mission



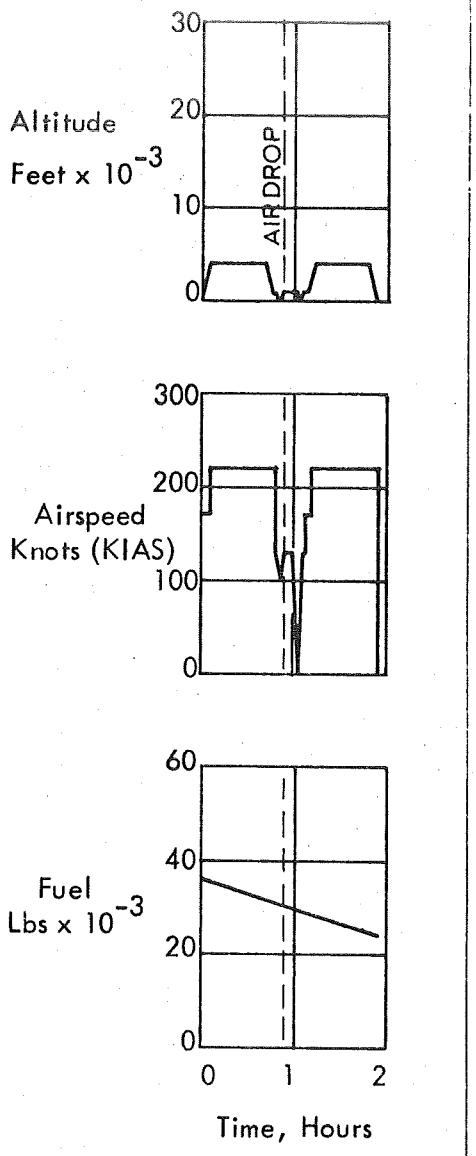
CARGO	1000 LBS
> 40	1%
30 - 40	4%
20 - 30	26%
10 - 20	51%
0 - 10	18%

Figure B-4. - Mission 4 - Short Range Logistic Mission



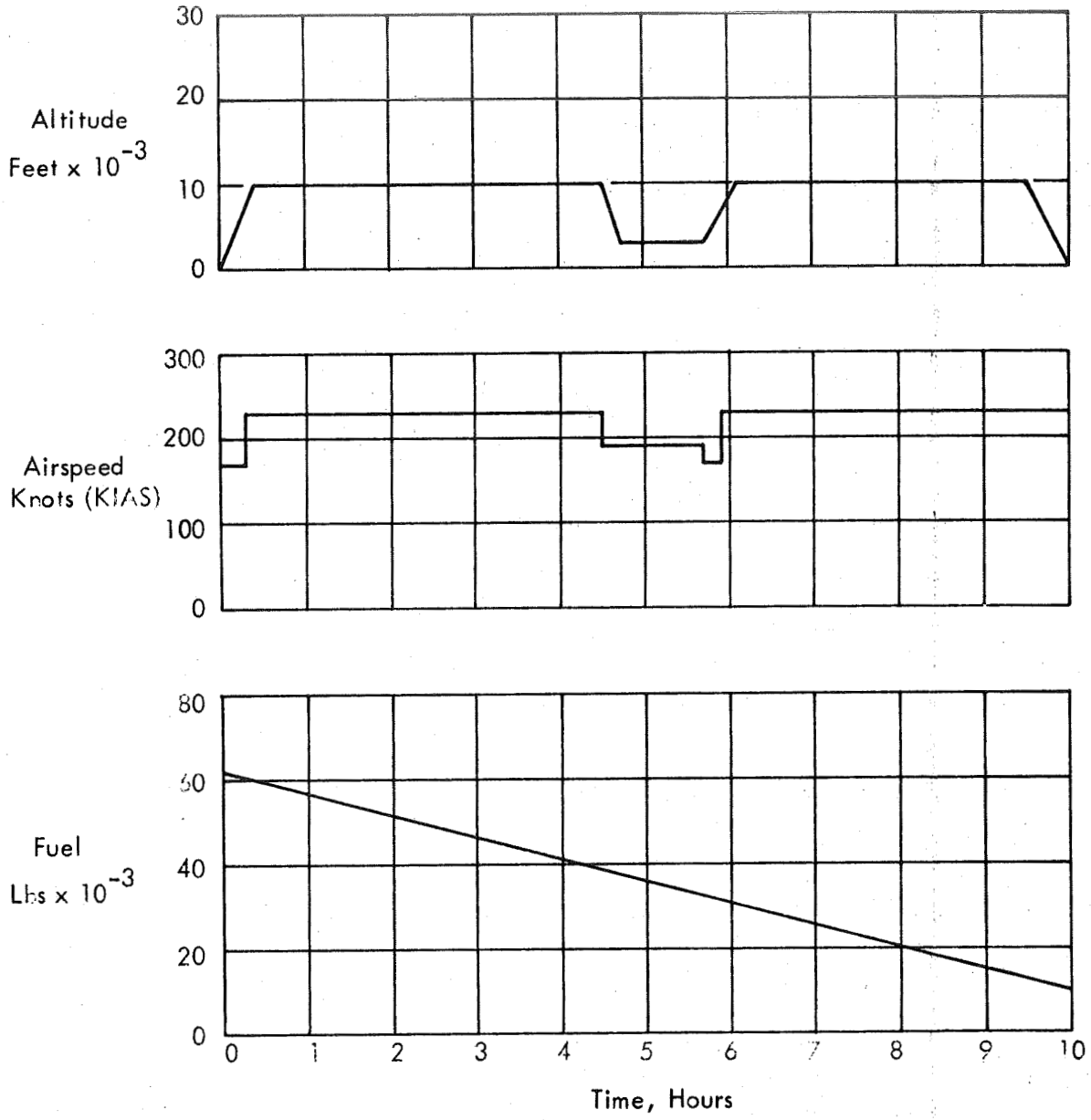
CARGO	1000 LBS
30 - 40	4%
20 - 30	28%
10 - 20	52%
0 - 10	16%

Figure B-5. - Mission 5 - Long Range Logistic Mission



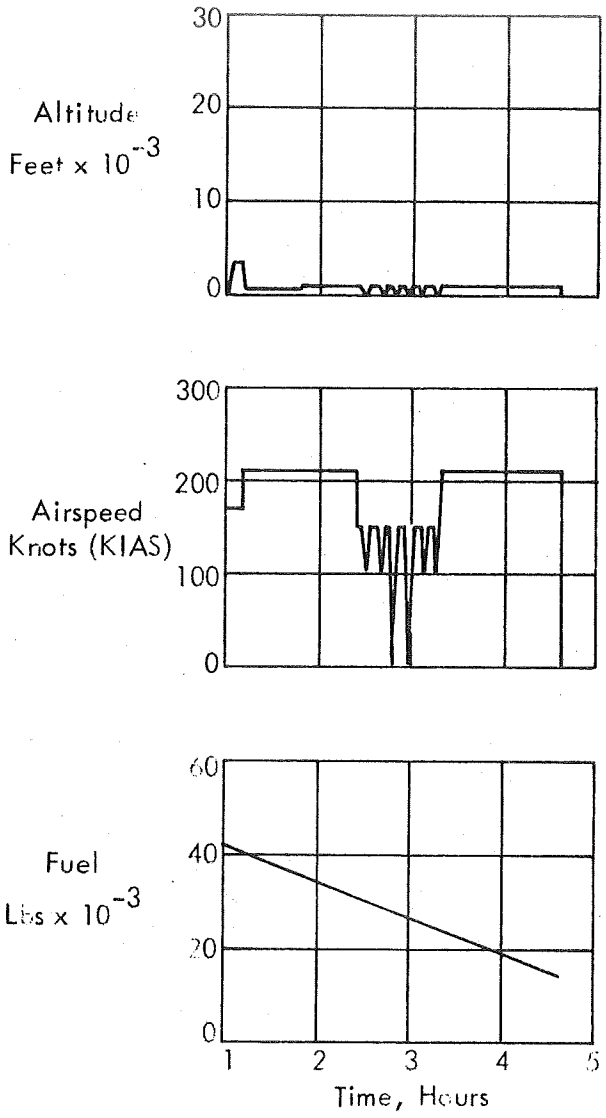
CARGO	1000 LBS	
	Before A/D	After A/D
>40	1%	
30 - 40	2%	
20 - 30	9%	
10 - 20	52%	14%
0 - 10	36%	86%

Figure B-6. - Mission 6 - Air Drop Mission



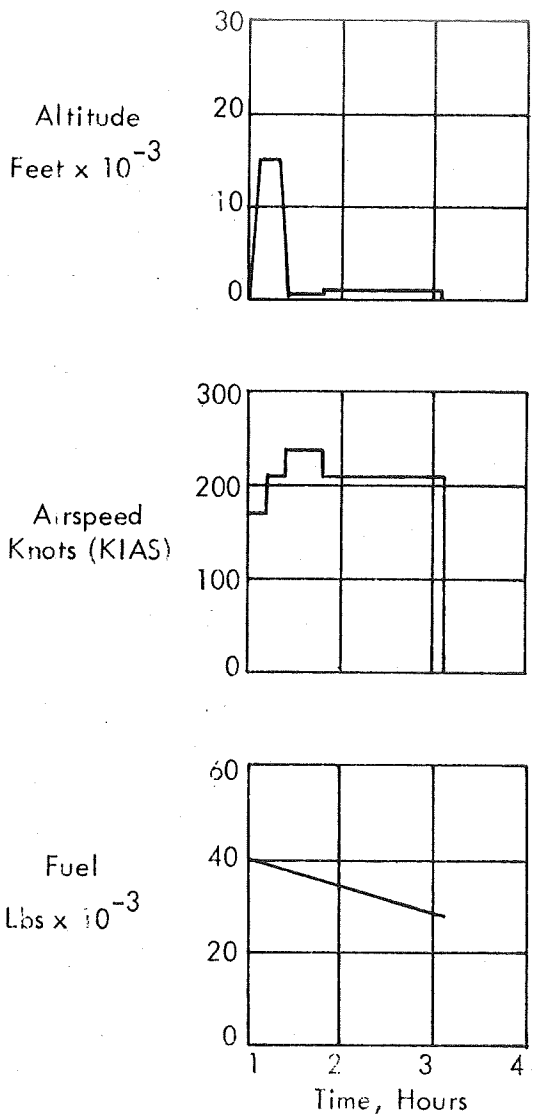
CARGO	1000 LBS
0 - 10	100%

Figure B-7. - Mission 7 - Storm Reconnaissance Mission



CARGO	1000 LBS
30 - 40	5%
20 - 30	2%
10 - 20	23%
0 - 10	70%

Figure B-8. - Mission 8 - Support Mission



CARGO	1000 LBS
30 - 40	1%
20 - 30	8%
10 - 20	24%
0 - 10	67%

Figure B-9. - Mission 9 - Low Level Mission

APPENDIX C

to

Part G

General Section

Strain Survey

STRAIN READINGS

In order to obtain a better definition of the stresses in the vicinity of the cracks, and to monitor the stresses during testing, strain measurements were taken during each crack propagation test and each residual strength test. This appendix contains the tabulation of the strains measured during the residual strength tests. The strains encountered during the crack propagation tests were commensurate within good accuracy with those of the residual strength tests.

Specimen No. 1

Pursuant to the contract work statement, five bonded electrical resistance type axial strain gages were applied to the lower (tension) surface. In addition, three Lockheed-funded bonded electrical resistance-type rosette strain gages were also applied to the lower surface. All such strain measurements (5 axial channels and 9 rosette channels) were recorded on a 30 channel automatic digital strain indicator. The locations of the axial and rosette strain gages is shown in Figure C-1. The readings obtained from the axial strain gages are given in Table C-I. It should be noted that all axial gage readings are given in terms of strain in micro-inches per inch and are measured in the spanwise direction. The readings and the solutions obtained from the rosette strain gages are given in Table C-II. The values given for principal stresses in these calculations are limited in their accuracy by the accuracy to which the strain gage readings may be obtained, which is on the order of 10 to 20 micro-inches with the Digital Strain Indicator.

In addition to strain gage readings on Specimen No. 1, a Lockheed-funded photoelastic strain survey was conducted in the area of the principal crack at W.S. 176L prior to the residual strength test. In this photoelastic survey, a birefringent plastic coating was bonded to the skin. With a load change the birefringent coating, as viewed through a polariscope, allowed direct observation of the principal stress difference, $S_p - S_q$. Through the polariscope, areas of constant value of shear stress $\frac{1}{2}(S_p - S_q)$ appear as bands of color, called fringes. Knowing the stress level required for producing one fringe gives a means of determining the stress in the area of panel skin over which the plastic is bonded. The areas in which the birefringent coatings were applied and the points at which stress readings were recorded are shown in Figure C-2. The recorded data from this photoelastic survey, which was conducted at 20% limit load, is given in Table C-III.

Specimens No. 2, 3, 4, 5, and 13

On each of Specimens 2, 3, 4, 5, and 13, only two axial strain gages were applied. One was at W.S. 176L (W.S. 176R for Specimen No. 3) on the centerline of Stringer 14 (Gage Nos. 2001, 3002, 4001, 5001 and 13001), and another at W.S. 0, 2.64 inches aft of the Stringer 16 centerline (Gage Nos. 2003, 3001, 4003, 5003 and 13003). On Specimens Number 2 and 3 the strains were recorded using a digital strain indicator which recorded strains only at particular moments during testing, but on Specimens 4, 5, and 13 strains were recorded continuously during both the crack propagation tests and the residual strength tests. The strains recorded during the residual strength tests are tabulated in Tables C-IV through C-VIII. The strains recorded during the crack propagation tests on Specimens 4, 5, and 13 were the same as those encountered at the corresponding load during the residual strength test.

Specimen No. 6

Specimen No. 6 was the first specimen tested in downbending, and as such, five axial strain gages were applied to the upper surface and upper surface stiffeners, as shown in Figure C-3. The strain readings from these five gages were recorded on an eight channel continuous recorder and these readings are tabulated in Table C-IX.

Specimens Nos. 7, 8, 9, 10, 11, and 12

In each of these specimens, only two axial strain gages were applied, both externally. These were at W.S. 150L, 3.45" forward of the Stringer 4 centerline (Gage Nos. 7001, 8001, 9001, 10001, 11001, and 12001), and at W.S. 61.6L, 3.45" forward of the Stringer 4 centerline (Gage Nos. 7003, 8003, 9003, 10003, 11003, and 12003). These gages were recorded on a continuous recorder, and this data is tabulated in Tables C-X thru C-XV.

Specimen 14

In order to obtain a good definition of the stresses introduced into the specimen near the W.S. 220 wing joint fitting by the steel test fixture, several additional strain gages were applied in that area. The locations of strain gages on Specimen No. 14 are shown in Figure C-4. In order to ascertain the effect of the chordwise shear application point, load conditions were applied which simulated a wide variation of the chordwise shear application point, prior to the crack propagation test. These conditions are given in Table C-XXVIII. Also, prior to the crack propagation test, each of the three load conditions

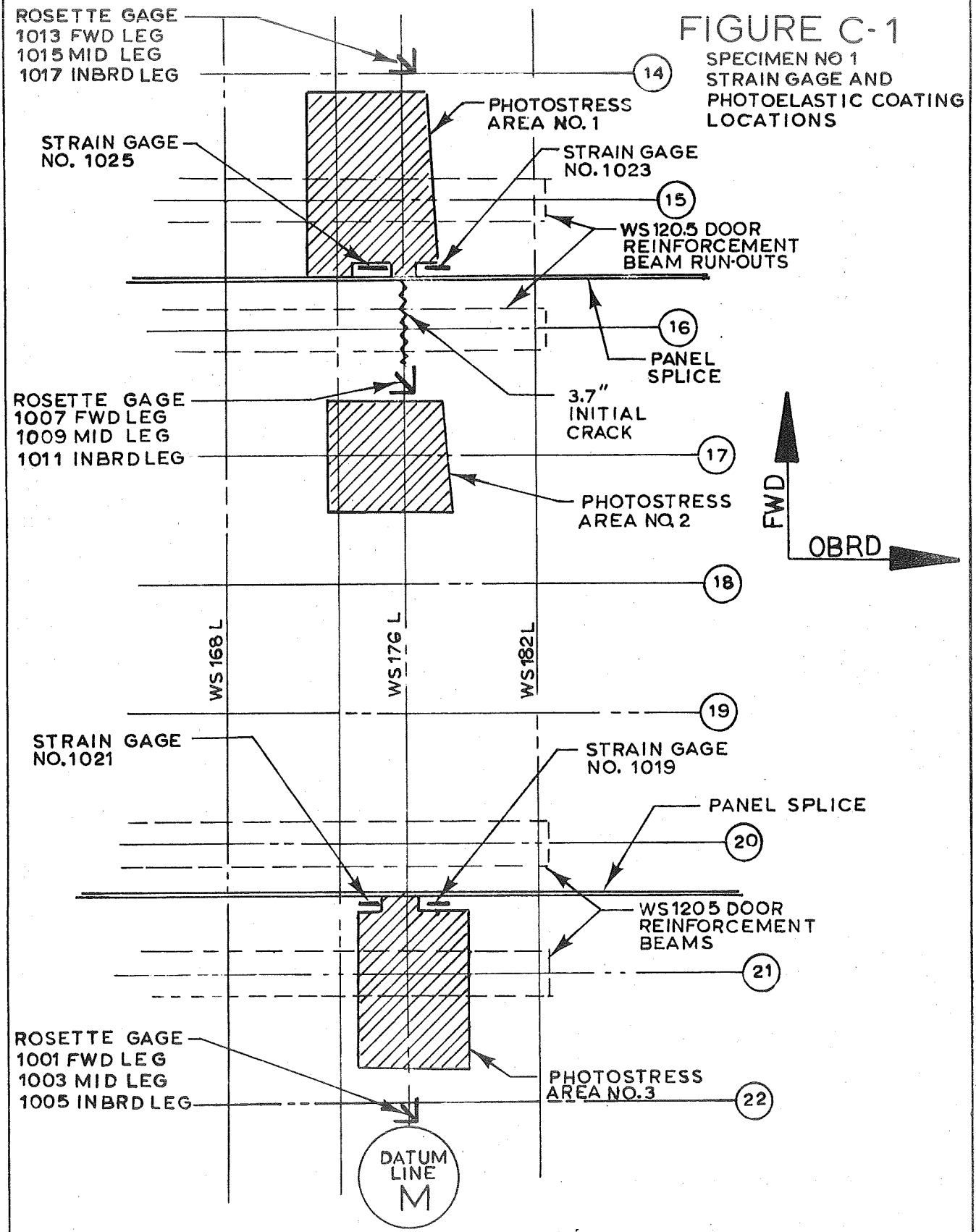
which were previously used in the program were applied and the strain gages read. These strain gage readings are tabulated in Tables C-XVI thru C-XX . Two rectangular rosette strain gages were applied to the forward and aft spar webs (which in this area are steel) as shown in Figure C-4. The readings and the solutions for these two rosette gages are given in Table C-XXI thru C-XXV,

The results of this stress distribution investigation proved quite satisfactory, as correlation with similar investigations performed on previous tests on the C-130 where an actual outer wing was used proved to be good.

Two gages, one at W.S. 176L on the centerline of Stringer 14, and one at W.S. 0, 2.64 inches aft of the centerline of Stringer 16, were monitored continuously during the crack propagation tests, and all gages were read during the residual strength test. The tabulated readings and rosette solutions from the residual strength test are given in Tables XXVI and XXVII, respectively

FIGURE C-1

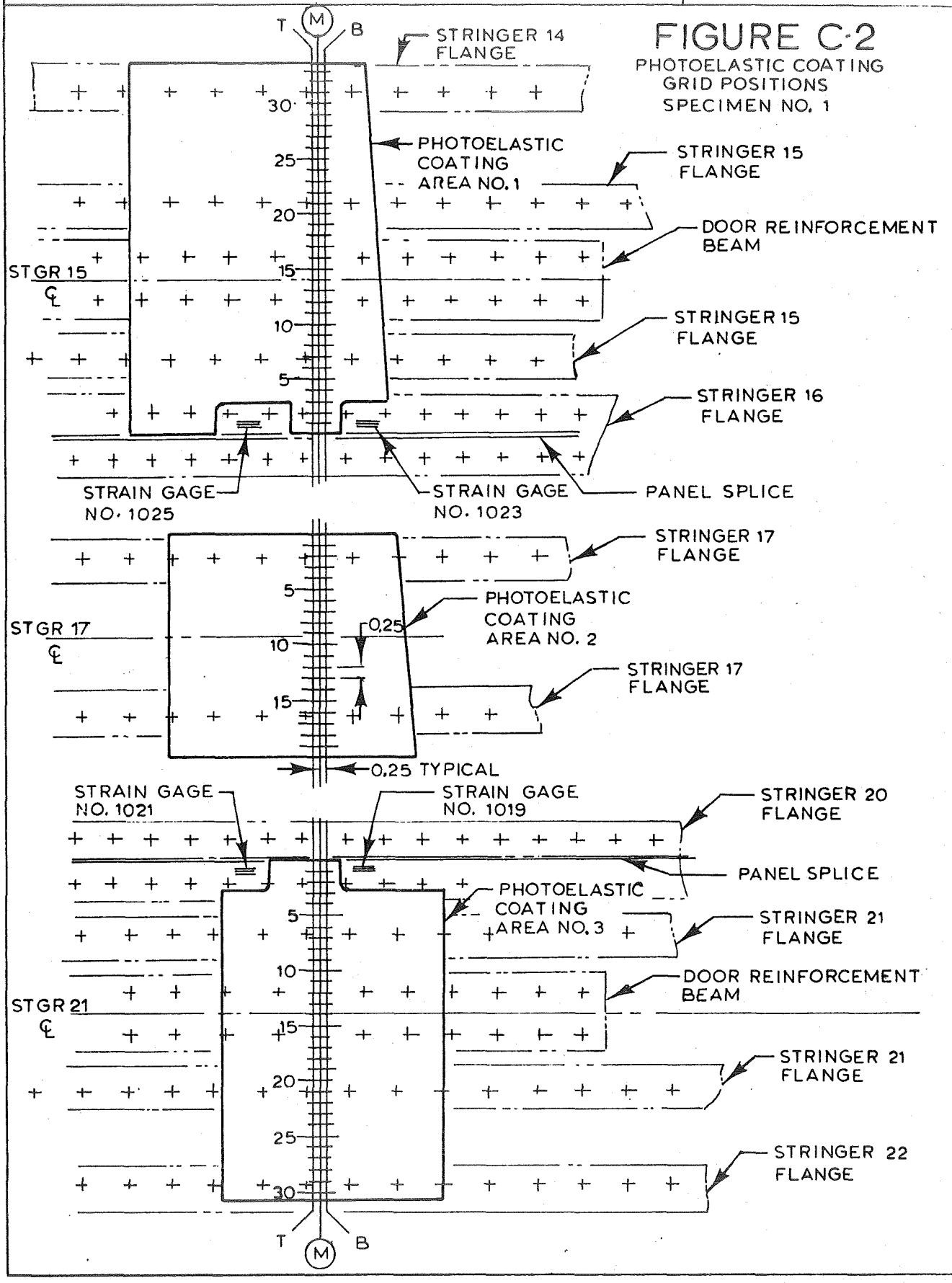
SPECIMEN NO 1
STRAIN GAGE AND
PHOTOELASTIC COATING
LOCATIONS



AXIAL STRAIN GAGE NO. 1027 LOCATED WS 0, 2.64" AFT OF THE STRINGER 16 CENTERLINE

FIGURE C-2

PHOTOELASTIC COATING GRID POSITIONS SPECIMEN NO. 1



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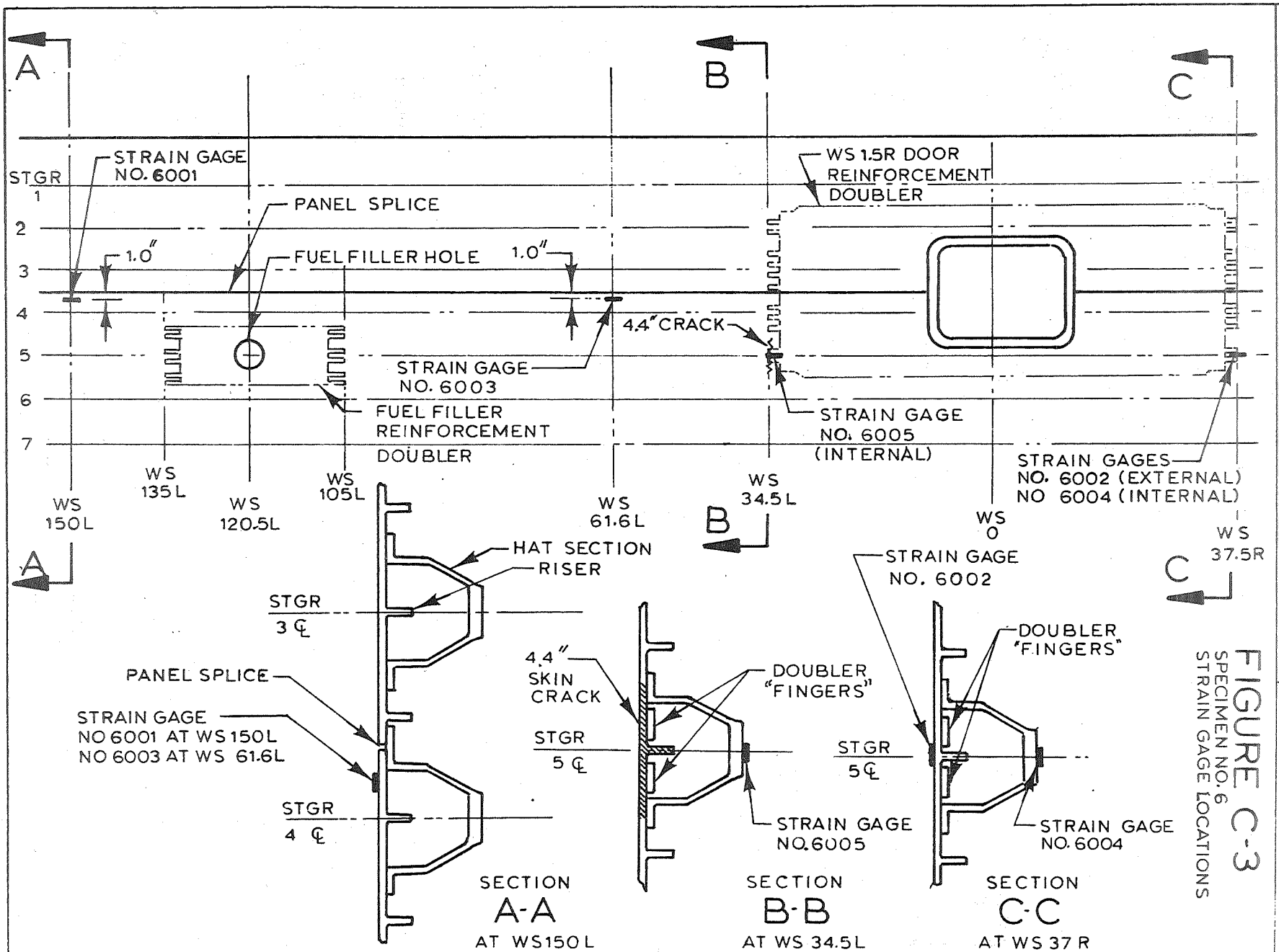

FIGURE C-3
SPECIMEN NO. 6
STRAIN GAGE LOCATIONS

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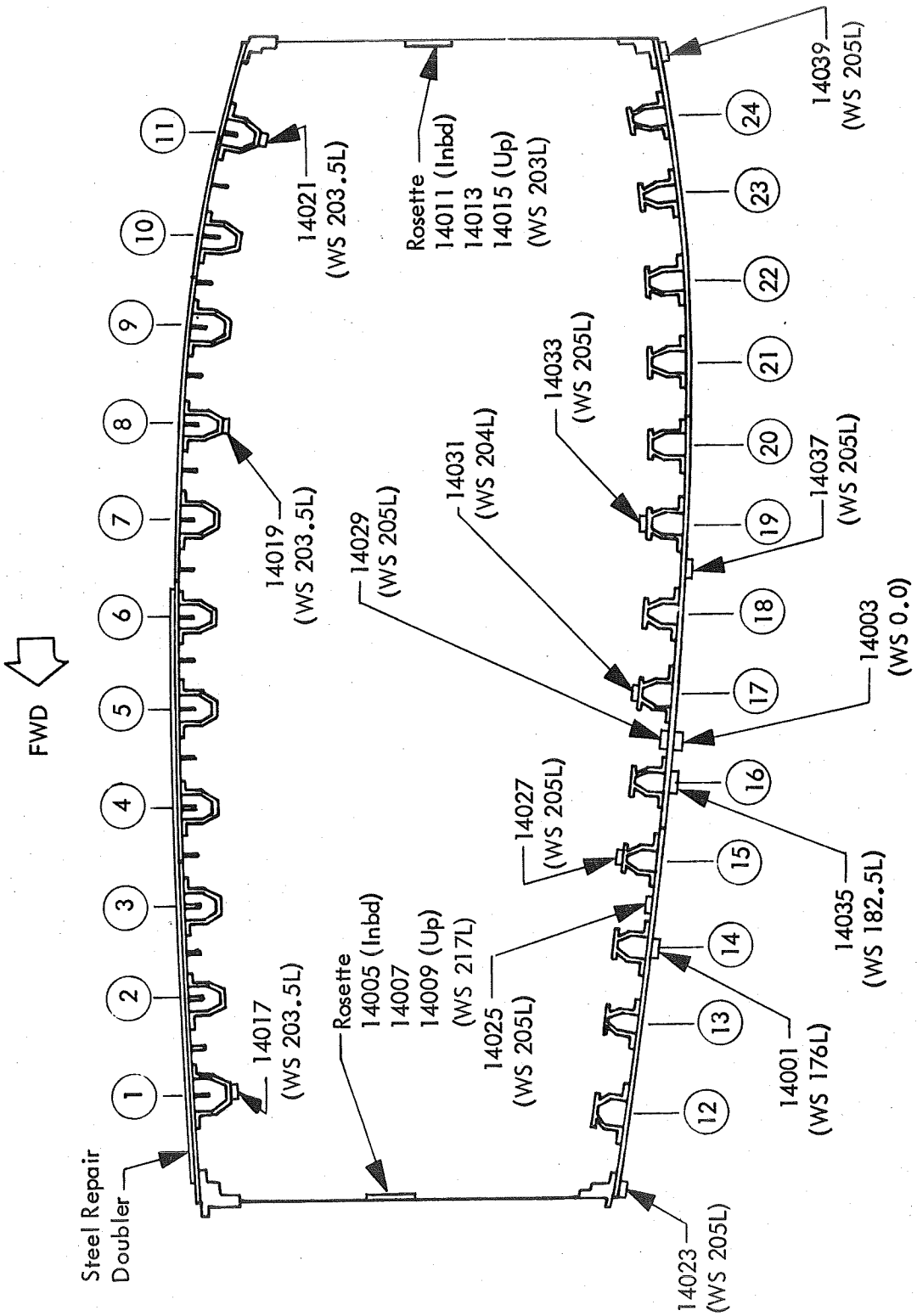

Figure C-4 - Strain gage locations, Specimen No. 14

TABLE C-I

READINGS FROM AXIAL STRAIN GAGES,
RESIDUAL STRENGTH TEST, SPECIMEN NO. 1

Percent Limit Load	Strain Readings In Microinches Per Inch					Comments
	Gage No. 1019	Gage No. 1021	Gage No. 1023	Gage No. 1025	Gage No. 1027	
0	0	0	0	0	0	
10	270	300	350	350	270	
20	530	600	700	700	630	
30	830	1000	1030	1100	1000	
40	1200	1400	1350	1500	1400	
50	1400	1730	1630	1800	1700	
60	1630	2030	1930	2230	2100	
70	2000	2400	2330	2700	2470	
80	2230	2770	2770	3100	2900	
87	2430	3000	3000	3400	3100	Before noise
87	2970	3800	3000	3400	3100	After noise
90	3200	4000	3170	3570	3230	
100	3600	4600	3600	4100	3600	
20	1030	1200	770	830	700	
50	2000	2570	1900	2100	1730	
70	2700	3400	2530	2900	2470	
100	3700	4600	3600	4070	3570	
110	4270	5330	4070	4500	4000	
-	4370	5430	4100	4600	4000	20 seconds to failure
	4400	5530	4200	4700	4050	10 seconds to failure
112.1	4400	5570	4200	4700	4100	Prior to occurance @ 7-1/2 sec.
112.1	4600	5600	4200	4700	4100	After occurance @ 7-1/2 sec.
	4650	5600	4250	4700	4170	5 seconds to failure
115.9	4700	5750	4300	4800	4170	Prior to failure
115.9	*	-400	*	0	300	After failure

* 'Pegged' positive

TABLE C-II

READINGS AND SOLUTIONS FROM ROSETTE STRAIN GAGES,
RESIDUAL STRENGTH TEST, SPECIMEN NO. 1

Percent Limit Load	Indicated Readings (μ In/In) Channel 1001	Channel 1003	Channel 1005	Rosette Solution* Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 1001	Max Shear Stress
10	-70	+200	+350	358.6	-79	3844	455	82.0°	1694
20	-50	-440	+740	764	-175	8163	891	81.0°	3635
30	-250	+640	+1100	1134	-287	12013	1008	81.2°	5502
40	-350	+860	+1500	1544	-398	16329	1287	81.4°	7520
50	-460	+1080	+1860	1922	-526	20210	1251	80.9°	9479
60	-570	+1320	+2240	2323	-658	24341	1255	80.5°	11543
70	-650	+1560	+2620	2720	-756	28556	1637	80.3°	13460
80	-770	+1810	+3020	3142	-899	32888	1593	80.1°	15647
90	-770	+2180	+3500	3653	-931	38673	3173	79.6°	17750
100	-870	+2490	+3920	4110	-1069	43429	3321	79.1°	20054
20	-140	+540	+860	892	-174	9646	1391	80.1°	4128
50	-380	+1260	+1960	2052	-477	21899	2314	79.0°	9793
70	-560	+1750	+2760	2885	-691	30711	3017	79.3°	13846
100	-830	+2420	+3790	3977	-1026	42056	3311	78.9°	19373
110	-930	+2780	+4350	4562	-1152	48337	4086	79.0°	22125
112	-950	+2820	+4460	4666	-1166	49486	4320	79.2°	22583

* Permanent Strain Set is not compensated for in the Rosette Solution.

Parameters Used
E = Young's Modulus = 10.3×10^6 PSI
= Poission's Ratio = 0.330

TABLE C-II, Continued

READINGS AND SOLUTIONS FROM ROSETTE STRAIN GAGES,
RESIDUAL STRENGTH TEST, SPECIMEN NO. 1

Percent Limit Load	Indicated Readings (μ In/In) Channel 1007	Channel 1009	Channel 1011	Rosette Solution* Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	Prin. CCW from Ch 1007	Max Shear Stress
10	-130	+230	+440	450	-141	4664	87	82.6°	2288
20	+50	+540	+940	942	45	11059	4113	87.1°	3473
30	+30	+800	+1400	1405	21	16320	5602	86.6°	5359
40	-30	+1040	+1860	1868	-43	21428	6628	86.3°	7400
50	-50	+1300	+2280	2295	-71	26257	7933	85.2°	9161
60	-70	+1580	+2720	2743	-101	31320	9295	84.8°	11012
70	-30	+1880	+3160	3191	-70	36617	11362	84.4°	12627
80	+70	+2260	+3720	3756	23	43502	14592	84.3°	14455
90	+210	+2700	+4300	4347	149	50814	18303	83.8°	16255
100	+520	+3360	+5220	5269	453	62631	25334	84.1°	18648
20	+550	+1050	+1450	1451	542	18839	11799	86.8°	3519
50	+630	+1970	+2900	2917	603	36016	18096	84.8°	8960
70	+660	+2560	+2840	3868	618	47067	21897	84.7°	12584
100	+550	+3300	+5100	5148	485	61354	25242	84.1°	18056
110	+690	+4050	+6180	6246	603	74496	30794	83.7°	21850
112	+730	+4190	+6400	6466	643	77192	32096	83.8°	22548

* Permanent Strain Set is not compensated for in the Rosette Solution.

Parameters Used

E = Young's Modulus = 10.3×10^6 PSI
= Poission's Ratio = 0.330

TABLE C-II, Continued

READINGS AND SOLUTIONS FROM ROSETTE STRAIN GAGES,
RESIDUAL STRENGTH TEST, SPECIMEN NO. 1

Percent Limit Load	Indicated Readings (μ In/In) Channel 1013	Channel 1015	Channel 1017	Rosette Solution* Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 1013	Max Shear Stress
10	-40	+50	+310	329	-61	3570	550	102.9°	1510
20	-120	+100	+690	731	-162	7832	916	102.3°	3458
30	-190	+150	+980	1030	-242	10982	1132	101.4°	4925
40	-240	+180	+1360	1447	-330	15467	1705	102.7°	6880
50	-320	+220	+1700	1805	-429	19227	1926	102.5°	8650
60	-380	+260	+2020	2146	-511	22856	2279	102.5°	10288
70	-440	+320	+2380	2524	-590	26923	2807	102.4°	12058
80	-510	+350	+2740	2913	-690	31039	3136	102.6°	1395
90	-570	+400	+3060	3249	-766	34633	3539	102.5°	15547
100	-650	+470	+3460	3665	-864	39067	3993	102.3°	17537
20	-170	+100	+780	822	-214	8685	661	101.7°	4011
50	-340	+260	+1780	1877	-441	20013	2062	101.7°	8976
70	-460	+340	+2420	2557	-603	27256	2783	102.0°	12236
100	-660	+460	+3400	3597	-865	38277	3722	102.1°	17277
110	-750	+500	+3840	4070	-989	43272	4093	102.3°	19589
112	-760	+520	+3900	4128	-998	43908	4210	102.2°	19848

* Permanent Strain Set is not compensated for in the Rosette Solution.

Parameters Used

E = Young's Modulus = 10.3×10^6 PSI
= Poission's Ratio = 0.330

TABLE C-III

READINGS AND SOLUTIONS FROM PHOTOELASTIC SURVEY
RESIDUAL STRENGTH TEST, SPECIMEN NO. 1 (c)
(20% LIMIT LOAD)

Coating No. 1 (b)

Grid Position		Angle (a)	$\sigma_x - \sigma_y$ (d)
0	T	90°	11194
	M	↑	9341
	B	↑	8646
3	T	↑	10808
	M	↑	9418
	B	↑	9341
4	T	↑	8492
	M	↑	9032
	B	↑	9110
5	T	↑	8724
	M	↑	8569
	B	↑	8878
6	T	↑	9650
	M	↑	8955
	B	↑	9264
9	T	↑	6408
	M	↑	6408
	B	↑	6408
10	T	↑	7720
	M	↑	7952
	B	↑	9264
14	T	↑	11445
	M	↑	12208
	B	90°	12971
18	T	110°	8393
	M	↑	10682
	B	↑	10682
19	T	↑	7935
	M	↑	8698
	B	↑	9156
22	T	↑	8012
	M	↑	↑
	B	↑	↑
23	T	↑	↑
	M	↑	↑
	B	110°	8012

Coating No. 1 (b)

Grid Position		Angle (a)	$\sigma_x - \sigma_y$ (d)
24	T	110°	8012
	M	↓	8190
	B	↓	↓
25	T	↓	↓
	M	↓	↓
	B	↓	↓
26	T	↓	↓
	M	↓	↓
	B	↓	↓
27	T	↓	↓
	M	↓	↓
	B	↓	↓
28	T	↓	↓
	M	↓	↓
	B	↓	↓
29	T	↓	↓
	M	↓	↓
	B	↓	8190
33	T	↓	7020
	M	↓	7020
	B	110°	7020

Coating No. 2

Grid Position		Angle (a)	$\sigma_x - \sigma_y$ (d)
0	T	90°	9744
	M	↑	10920
	B	↑	10080
1	T	↑	8904
	M	↑	8904
	B	↑	8400
4	T	↑	8148
	M	↑	7896
	B	↑	7476
5	T	↑	7560
	M	↑	7560
	B	90°	7560

Coating No. 2 (b)

Grid Position		Angle (a)	$\sigma_x - \sigma_y$ (d)
6	T	90°	7560
	M	↑	↓
	B	↑	↓
7	T	↑	↓
	M	↑	↓
	B	↑	↓
8	T	↑	↓
	M	↑	↓
	B	↑	↓
9	T	↑	↓
	M	↑	↓
	B	↑	↓
10	T	↑	↓
	M	↑	↓
	B	↑	↓
11	T	↑	↓
	M	↑	↓
	B	↑	↓
12	T	↑	↓
	M	↑	↓
	B	↑	↓
13	T	↑	↓
	M	↑	↓
	B	↑	↓
14	T	↑	↓
	M	↑	↓
	B	↑	7560
15	T	↑	↓
	M	↑	↓
	B	↑	7560
18	T	↑	8400
	M	↑	↑
	B	↑	↑
19	T	↑	↑
	M	↑	↑
	B	90°	8400

TABLE C-III, Continued

READINGS AND SOLUTIONS FROM PHOTOELASTIC SURVEY RESIDUAL STRENGTH TEST SPECIMEN NO. 1 (c) (20% LIMIT LOAD)

Coating No. 3 (b)				Coating No. 3 (b)			
Grid Position		Angle (a)	$\sigma_p - \sigma_q$ (d)	Grid Position		Angle (a)	$\sigma_p - \sigma_q$ (d)
0	T	90°	7793	20	T	90°	6216
	M		7710		M		
	B		7212		B		
1	T		8539	21	T		
	M		7378		M		
	B		6218		B		6216
2	T		8290	22	T		8820
	M		5886		M		
	B		5886		B		
3	T		8787	23	T		
	M				M		
	B				B		8820
4	T			24	T		8925
	M				M		
	B				B		
5	T			25	T		
	M				M		
	B				B		8925
6	T			26	T		8500
	M				M		
	B		8787		B		
9	T		5880	27	T		
	M		5880		M		
	B		5880		B		
10	T		7560	28	T		
	M		7560		M		
	B		8400		B	90°	8500
19	T		3360				
	M		5880				
	B	90°	7560				

(a) Angle to Max Principle Stress is measured so that the forward direction is 0°, the inboard direction 90°
(b) Grid positions at which fastener interference prevented a reading are deleted
(c) All measurements taken at 20% of limit upbending load
(d) Principle Stress Difference - 2 x Max Shear Stress

TABLE C-IV

READINGS FROM AXIAL STRAIN GAGES,
RESIDUAL STRENGTH TEST,
SPECIMEN NO. 2

Percent Limit Load	Strain Gage Readings Gage No. 2001	Gage No. 2003	Comments
0	0	0	
10	380	380	
20	740	760	
30	1090	1140	
40	1440	1540	
50	3400	1960	
60	2120	2340	
70	1390	2740	
50	1780	1960	
80	2290	3100	
90	3290	3580	
100	3660	3930	
20	820	840	
50	1940	1940	
80	3080	3080	
100	2870	3920	
110	5780	4300	
50	2140	2020	
80	3020	3130	
100	3760	3920	
110	4120	4330	
0 Failure	-1960	360	Failure at 114.5%

TABLE C-V

READINGS FROM AXIAL STRAIN GAGES,
RESIDUAL STRENGTH TEST,
SPECIMEN NO. 3

Percent Limit Load	Strain Gage Readings Gage No. 3001	Gage No. 3002	Comments
10	270	330	
20	640	720	
30	990	1140	
40	1370	1540	
50	1680	1970	
20	750	840	
50	1630	1940	
60	2070	2360	
70	2450	2800	
80	2800	3120	
90	3130	3550	
100	3560	3880	
20	830	840	
50	1850	1960	
100	3750	3880	
110	Gage has failed	4260	
After Failure	Gage has failed	Gage has failed	

TABLE C-VI

READINGS FROM AXIAL STRAIN GAGES
RESIDUAL STRENGTH TEST
SPECIMEN NO. 4

Percent Limit Load	Strain Gage Readings Gage No. 4001	Gage No. 4003	Comments
10	220	340	
20	600	700	
30	1060	1060	
40	1460	1440	
50	1900	1780	
Occur-rence	2100	2000	
60	2280	2260	
20	700	840	
60	2280	2260	
64.8	2480	2380	Immed. prior to failure

TABLE C-VII

READINGS FROM AXIAL STRAIN GAGES
RESIDUAL STRENGTH TEST
SPECIMEN NO. 5

Percent Limit Load	Strain Gage Readings Gage No. 5001	Gage No. 5003	Comments
10	268	384	
20	614	720	
30	934	1120	
40	1280	1500	
50	1600	1896	
60	1964	2270	
20	640	800	
60	1956	2240	
70	2326	2570	
20	640	704	
70	2234	2502	
80	2956	2880	
20	680	608	
80	2946	2766	
86.8	3126	3010	Immed. prior to failure

TABLE C- VIII

READINGS FROM AXIAL STRAIN GAGES, RESIDUAL STRENGTH TEST SPECIMEN NO. 13

Percent Limit Load	Gage No. 13001	Gage No. 13003	Comments
10	326	306	
20	720	680	
30	1090	1040	
40	1450	1380	
42.7	1580	1500	Pop
20	750	710	
40	1460	1390	
44.4	1640	1550	Pop
20	790	730	
40	1460	1380	
50	1830	1720	
52.3	1910	1790	Pop
20	800	720	
50	1830	1710	
54.2	1980	1820	Pop
20	800	710	
50	1810	1690	
59	2160	1940	Pop
20	800	686	
50	1840	1640	
60	2210	1970	
62.3	2450	Gage out	Loud bang, large deflection as in failure
0	10	Gage out	
20	680	Gage out	
30	1020	Gage out	
40	1380	Gage out	
50	1700	Gage out	
52.3	1740	Gage out	
0	50	Gage out	

TABLE C- IX

READINGS FROM AXIAL STRAIN GAGES, RESIDUAL STRENGTH TEST, SPECIMEN NO. 6

Percent Limit Load	Strain Readings In Microinches Per Inch					Comments
	Gage No. 1019	Gage No. 1021	Gage No. 1023	Gage No. 1025	Gage No. 1027	
10	320	380	390	380	380	
20	650	810	755	690	710	
30	936	1070	1120	1000	1000	
40	1250	1560	1500	1310	1370	
50	1610	2030	1920	1720	1750	
60	1890	2470	2280	2040	2060	
70	2220	2880	2680	2380	2460	
20	660	940	840	750	780	
50	1590	2060	1940	1730	1750	
70	2220	2880	2680	2360	2420	
76	2620	3380	3000	2750	2750	
40	1400	1750	1600	1600	1400	
70	2350	4000	2750	2800	2500	
80	2700	4450	3150	3200	2800	
	3050	5030	3450	3500	3150	Prior to occurrence
	3050	5450	3450	3500	3150	After occurrence
90	3100	-	3500	3550	3200	Strain gage 2 has failed
20	750	-	970	970	800	
50	1600	-	1950	2000	1700	
90	3100	-	3550	3600	3200	
96.3	3200	-	3650	3700	3300	Immediately prior to failure

TABLE C-X

READINGS FROM AXIAL STRAIN GAGES
RESIDUAL STRENGTH TEST
SPECIMEN NO. 7

Percent Limit Load	Strain Gage Readings Gage No. 7001	Gage No. 7003	Comments
5.5	150	200	
11.1	300	430	
16.6	440	650	
22.2	600	850	
27.7	750	1100	
33.3	950	1400	
38.8	1100	1600	
44.4	1250	1900	
49.9	1400	2100	
55.5	1600	2350	
11.1	350	600	
55.5	1600	2400	
61.1	1750	2600	
66.6	1950	2900	
72.2	2150	3200	
11.1	400	600	
50	1500	2050	
70	2100	2900	
80	2450	3350	
90	2750	3800	
97.7	2900	4100	Immed. prior to failure

TABLE -XI

READINGS FROM AXIAL STRAIN GAGES
RESIDUAL STRENGTH TEST
SPECIMEN NO. 8

Percent Limit Load	Strain Gage Readings Gage No. 8001	Gage No. 8003	Comments
0	0	80	
10	160	380	
20	520	780	
30	900	1200	
40	1250	1550	
50	1600	1950	
60	1980	2350	
70	2350	2750	
80	2700	3200	
90	3050	3550	
100	3400	3950	
20	550	850	
0	0	100	
50	1550	1900	
90	3000	3550	
100	3400	3950	
105	3550	4200	Small sonic report
20	500	850	
90	3000	3550	
100	3400	4000	
110	3800	4450	
116.7	3900	4600	Immed. prior to failure
0	0	80	Subsequent to failure

TABLE C-XII

READINGS FROM AXIAL STRAIN GAGES
RESIDUAL STRENGTH TEST
SPECIMEN NO. 9

Percent Limit Load	Strain Gage Readings Gage No. 9001	Gage No. 9003	Comments
10	230	280	
20	500	640	
30	920	1120	
40	1200	1490	
50	1480	1850	
60	1750	2000(+)	Zero setting disturbed on 9003
70	2060	2370(+)	
75	2220	2720	
20	620	780	
40	1190	1450	
70	2120	2600	
78	2360	2880	
20	630	750	
40	1200	1500	
70	2130	2650	
80	2440	3040	
81	2460	3060	Immed. prior to failure

TABLE C-XIII

READINGS FROM AXIAL STRAIN GAGES
RESIDUAL STRENGTH TEST
SPECIMEN NO. 10

Percent Limit Load	Strain Gage Readings Gage No. 10001	Gage No. 10003	Comments
10	270	320	
20	600	710	
30	930	1100	
40	1290	1520	
50	1650	1940	
60	2000	2360	
70	2340	2780	
76	2560	3040	
20	640	800	
50	1620	1940	
70	2340	2780	
80	2710	3220	
84	2880	3580	
20	600	840	
60	1980	2490	
80	2700	3370	
88	3100	3800(+)	10003 Ran off scale
20	660	960	
60	2020	2610	
80	2760	3510	
87	3000	3690	

TABLE C- XIV

READINGS FROM AXIAL STRAIN GAGES
RESIDUAL STRENGTH TEST
SPECIMEN NO. 11

Percent Limit Load	Strain Gage Readings Gage No. 11001	Gage No. 11003	Comments
10	294	330	
20	640	700	
30	1000	1090	
40	1360	1490	
50	1700	1870	
57.3	1940	2120	Immed. prior to failure

TABLE C -XV

READINGS FROM AXIAL STRAIN GAGES
RESIDUAL STRENGTH TEST
SPECIMEN NO. 12

Percent Limit Load	Strain Gage Readings Gage No. 12001	Gage No. 12003	Comments
10	312	Inoperative	Gage 12003 inoperative throughout test.
20	652		
30	980		
40	1340		
50	1960		
20	660		
40	1260		
50	1580		
20	640		
50	1492		
56.4	1680		Immed. prior to failure
0	-160	Inoperative	

TABLE C-XVI

READINGS FROM AXIAL STRAIN GAGES, STRAIN SURVEY, TEST CONDITION 1, SPECIMEN NO. 14

Strain Readings In Microinches Per Inch

Percent Limit Load	Gage No. 14017	Gage No. 14019	Gage No. 14021	Gage No. 14023	Gage No. 14025	Gage No. 14027	Gage No. 14029	Gage No. 14031	Gage No. 14033	Gage No. 14035	Gage No. 14037	Gage No. 14039
20	-200	-310	-260	+220	+290	+310	+310	+310	+300	+340	0	+240
40	-340	-650	-540	+370	+690	+700	+720	+690	+700	+780	+440	+600
60	-720	-990	-820	+600	+1000	+1040	+1080	+1020	+1000	+1140	+760	+1060
80	-930	-1300	-1060	+800	+1330	+1380	+1420	+1390	+1380	+1540	+1140	+1180
100	-1170	-1650	-1330	+1040	+1700	+1780	+1800	+1810	+1740	+1960	+1540	+1530
20	-200	-330	-260	+140	+340	+320	+380	+300	+300	+870	+40	+320
0	+20	-20	-20	-100	+20	0	+40	+10	-20	+40	-300	+80

TABLE C-XVII

READINGS FROM AXIAL STRAIN GAGES, STRAIN SURVEY, TEST CONDITION 2, SPECIMEN NO. 14

Strain Readings In Microinches Per Inch

Percent Limit Load	Gage No. 14017	Gage No. 14019	Gage No. 14021	Gage No. 14023	Gage No. 14025	Gage No. 14027	Gage No. 14029	Gage No. 14031	Gage No. 14033	Gage No. 14035	Gage No. 14037	Gage No. 14039
20	-200	-300	-250	140	320	310	320	310	330	360	340	270
40	-440	-650	-510	360	660	680	700	670	700	740	700	580
60	-680	-980	-800	600	1000	1020	1060	1030	1040	1130	1070	890
80	-930	-1300	-1060	820	1340	1400	1430	1410	1410	1520	1460	1210
100	-1180	-1640	-1330	1040	1660	1740	1760	1790	1740	1880	1820	1540
0*	-20	-60	-50	60	40	20	40	30	40	40	40	20

* Due to equipment problems, 20% reading not taken 2nd time.

TABLE C-XVIII

READINGS FROM AXIAL STRAIN GAGES, STRAIN SURVEY
UPBENDING, SPECIMEN NO. 14

Strain Readings In Microinches Per Inch

Percent Limit Load	Gage No. 14017	Gage No. 14019	Gage No. 14021	Gage No. 14023	Gage No. 14025	Gage No. 14027	Gage No. 14029	Gage No. 14031	Gage No. 14033	Gage No. 14035	Gage No. 14037	Gage No. 14039
10	-220	-320	-140	230	380	400	400	390	380	440	340	280
20	-500	-680	-430	410	710	760	740	740	740	800	760	580
30	-740	-980	-710	650	1060	1140	1120	1110	1120	1240	1150	900
40	-900	-1260	-980	910	1440	1520	1500	1530	1520	1650	1560	1220
50	-1260	-1760	-1470	1040	1740	1820	1810	1820	1780	2000	1940	1520
10	-260	-380	-170	220	380	400	400	360	380	440	400	320
0	-20	-40	70	0	20	40	40	20	40	40	40	40

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TABLE C-XIX

READINGS FROM AXIAL STRAIN GAGES, STRAIN SURVEY, DOWNBENDING 1, SPECIMEN NO. 14

Strain Readings In Microinches Per Inch

Percent Limit Load	Gage No. 14017	Gage No. 14019	Gage No. 14021	Gage No. 14023	Gage No. 14025	Gage No. 14027	Gage No. 14029	Gage No. 14031	Gage No. 14033	Gage No. 14035	Gage No. 14037	Gage No. 14039
10	-260	-340	-260	-220	-330	-320	-320	-300	-300	-360	-340	-260
20	500	680	520	-430	-640	-680	-680	-600	-600	-760	-730	-560
30	700	970	740	-600	-960	-980	-1040	-880	-860	-1120	-1080	-820
40	900	1330	980	-760	-1280	-1260	-1380	-1160	-1160	-1480	-1420	-1060
50	1180	1670	1260	-970	-1610	-1560	-1740	-1480	-1470	-1920	-1840	-1340
10	240	350	240	-240	-380	-280	-400	-260	-260	-420	-440	-360
0	40	40	10	-60	-80	0	-80	0	20	-80	-100	-100

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TABLE C-XX

READINGS FROM AXIAL STRAIN GAGES, STRAIN SURVEY,
DOWNBENDING 2, SPECIMEN NO. 14

Strain Readings In Microinches Per Inch

Percent Limit Load	Gage No. 14017	Gage No. 14019	Gage No. 14021	Gage No. 14023	Gage No. 14025	Gage No. 14027	Gage No. 14029	Gage No. 14031	Gage No. 14033	Gage No. 14035	Gage No. 14037	Gage No. 14039
10	150	280	170	-140	-230	-260	-260	-280	-280	-290	-260	-140
20	370	580	400	-280	-500	-580	-560	-580	-580	-640	-560	-340
30	550	860	580	-380	-760	-870	-840	-840	-860	-980	-880	-490
10	180	300	180	-130	-260	-280	-270	-280	-260	-310	-300	-160
0	10	20	0	-20	-20	-20	-20	-20	-20	-20	-20	-70

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TABLE C-XXI

ROSETTE STRAIN GAGE SOLUTIONS
TEST CONDITION 1, SPECIMEN 14

Percent Limit Load	Indicated Readings (μIn/In) Channel 14005	Channel 14007	Channel 14009	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14005	Max Shear Stress
20	-60	-20	-80	-19	-121	-1906	-4136	39.4	1115
40	0	10	-140	37	-177	-696	-5346	24.4	2325
60	0	20	-240	65	-305	-1146	-9211	24.7	4033
80	40	20	-310	100	-369	-713	-10938	20.8	5113
100	80	40	-380	150	-449	52	-12998	19.8	6525
20	-20	-20	-110	-1	-129	-1413	-4197	22.5	1392
0	-40	-40	-20	-16	-44	-985	-1604	22.5	309

Percent Limit Load	Indicated Readings (μIn/In) Channel 14011	Channel 14013	Channel 14015	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14011	Max Shear Stress
20	-30	-80	-20	30	-80	129	-2287	42.4	1208
40	50	-160	20	231	-161	4788	-2767	-42.8	4277
60	80	-220	0	304	-224	7479	-4027	-40.6	5753
180	180	-340	0	530	-351	13492	-5724	-39.1	9608
100	140	-380	0	527	-387	12981	-6939	-40.6	9960
20	120	-180	40	344	-184	9205	-2301	-40.6	5753
0	0	-40	0	40	-40	875	-875	45.0	875

TABLE C-XXII

ROSETTE STRAIN GAGE SOLUTIONS
TEST CONDITION 2, SPECIMEN 14

Percent Limit Load	Indicated Readings (μIn/In) Channel 14005	Channel 14007	Channel 14009	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14005	Max Shear Stress
20	-10	20	-70	28	-107	-259	-3193	31.7	1467
40	-10	20	-130	39	-178	-655	-5386	28.2	2366
60	20	40	-250	92	-321	-467	-9458	24.5	4495
80	40	60	-330	132	-422	-218	-12296	24.0	6039
100	70	60	-410	164	-503	-66	-14606	21.9	7270
0	-30	-10	-10	6	-34	-554	-1172	-22.5	309

Percent Limit Load	Indicated Readings (μIn/In) Channel 14011	Channel 14013	Channel 14015	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14011	Max Shear Stress
20	20	-50	0	71	-51	1762	-899	-40.3	1330
40	200	-60	0	289	-90	8442	189	-29.0	4127
60	100	-100	0	209	-109	5616	-1300	-35.8	3458
80	140	-120	0	273	-133	7449	-1408	-34.9	4428
100	180	-160	0	356	-177	9695	-1927	-35.1	5811
0	0	-30	0	30	-30	656	-656	45.0	656

TABLE C-XXIII

ROSETTE STRAIN GAGE SOLUTIONS
UPBENDING

Percent Limit Load	Indicated Readings (μIn/In) Channel 14005	Channel 14007	Channel 14009	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14005	Max Shear Stress
10	40	20	-40	45	-45	978	-978	13.3	978
20	20	20	-140	54	-173	-115	-5064	22.5	2474
30	40	20	-200	77	-237	-36	-6869	19.9	3416
40	50	40	-280	113	-342	-12	-9914	21.6	4951
50	20	0	-80	29	-88	-19	-2570	15.5	1275
0	0	0	0	0	0	0	0	45.0	0
10	20	0	-80	29	-88	-19	-2570	15.5	1275

Percent Limit Load	Indicated Readings (μIn/In) Channel 14011	Channel 14013	Channel 14015	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14005	Max Shear Stress
10	40	-80	40	160	-81	4351	-898	45.0	2624
20	30	-220	0	251	-221	5797	-4503	-43.2	5150
30	40	-320	0	362	-322	8312	-6586	-43.3	7449
40	80	-420	20	522	-423	12458	-8142	-43.2	10300
50	60	-720	-40	744	-724	16434	-15571	-43.1	16003
10	20	-140	10	171	-141	4039	-2744	-44.1	3392
0	0	0	0	0	0	0	0	45.0	0

TABLE C-XXIV

ROSETTE STRAIN GAGE SOLUTIONS
DOWNBENDING 1

Percent Limit Load	Indicated Readings (μ In/In) Channel 14005	Channel 14007	Channel 14009	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14005	Max Shear Stress
10	0	0	70	85	-15	2593	428	22.5	1083
20	0	0	110	133	-23	4075	672	22.5	1701
30	0	-20	170	220	-51	6623	714	25.5	2955
40	0	-30	250	324	-75	9749	1039	25.6	4355
50	20	-50	310	425	-96	12792	1449	28.0	5672
10	-20	-20	50	65	-35	1730	-435	22.5	1083
0	0	0	-10	2	-12	-61	-371	22.5	155

Percent Limit Load	Indicated Readings (μ In/In) Channel 14011	Channel 14013	Channel 14015	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14011	Max Shear Stress
10	0	100	10	101	-91	2296	-1865	-43.5	2081
20	-20	220	0	221	-241	4603	-5466	-43.8	5035
30	-20	320	0	321	-341	6789	-7652	-44.1	7221
40	20	450	-40	452	-472	9650	-10513	43.1	10082
50	-40	550	0	552	-592	11611	-13337	-44.0	12474
10	0	130	0	130	-130	2843	-2843	45.0	2843
0	20	40	0	42	-22	1123	-260	35.8	692

TABLE C-XXV

ROSETTE STRAIN GAGE SOLUTIONS
DOWNBENDING 2

Percent Limit Load	Indicated Readings (μ In/In) Channel 14005	Channel 14007	Channel 14009	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14005	Max Shear Stress
10	-20	-20	80	101	-41	2841	-252	22.5	1547
20	0	-40	160	225	-65	6607	298	28.2	3154
30	0	-20	100	136	-37	4039	276	27.2	1881
10	0	-20	100	136	-37	4039	276	27.2	1881
0	0	0	0	0	0	0	0	45.0	0

Percent Limit Load	Indicated Readings (μ In/In) Channel 14011	Channel 14013	Channel 14015	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14011	Max Shear Stress
10	0	180	0	181	-181	3937	-3937	45.0	3937
20	-20	360	-20	361	-401	7448	-9174	45.0	8311
30	0	560	-20	562	-582	12036	-12899	44.5	12468
10	0	200	0	201	-201	4374	-4374	45.0	4374
0	0	0	0	0	0	0	0	45.0	0

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TABLE C-XXVI

READINGS FROM AXIAL GAGES, RESIDUAL STRENGTH TEST,
SPECIMEN NO. 14

Strain Readings In Microinches Per Inch

Percent Limit Load	Gage No. 14017	Gage No. 14019	Gage No. 14021	Gage No. 14023	Gage No. 14025	Gage No. 14027	Gage No. 14029	Gage No. 14031	Gage No. 14033	Gage No. 14035	Gage No. 14037	Gage No. 14039
10	-240	-340	-240	180	360	390	320	360	320	340	300	300
20	-440	-680	-590	460	740	810	640	740	720	670	660	640
30	-780	-1020	-810	680	1060	1160	910	1080	1060	980	960	980
40	-1020	-1380	-1110	940	1450	1630	1250	1520	1460	1320	1340	1310
50	-1250	-1720	-1380	1200	1800	2030	1530	1900	1840	1650	1660	1660
50*	-960	-1820	-1380	1060	1820	2030	1550	1900	1840	1660	1660	1680
0	40	40	20	-40	0	-10	-10	0	-40	0	-20	-20

* After dropping load to 20%

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TABLE C-XXVII

ROSETTE STRAIN GAGE SOLUTIONS
RESIDUAL STRENGTH TEST, SPECIMEN 14

Percent Limit Load	Indicated Readings (μIn/In) Channel 14005	Channel 14007	Channel 14009	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14005	Max Shear Stress
10	40	0	-60	41	-61	684	-1547	5.7	1115
20	20	20	-150	56	-185	-176	-5434	22.5	2629
30	40	20	-220	81	-261	-160	-7608	20.1	3724
40	60	60	-280	132	-351	511	-10005	22.5	5258
50	80	80	-350	170	-440	824	-124 6	22.5	6650
50	80	70	-380	170	-469	488	-13434	21.9	6961
0	0	0	-20	4	-24	-122	-741	22.5	309

Percent Limit Load	Indicated Readings (μIn/In) Channel 14011	Channel 14013	Channel 14015	Rosette Solution Prin. Strain	Prin. Strain	Prin. Stress	Prin. Stress	CCW from Ch 14011	Max Shear Stress
10	30	-100	0	131	-101	3184	-1889	-41.3	2536
20	50	-210	0	262	-212	6247	-4090	-42.0	5169
30	80	-340	-10	414	-344	9771	-6750	-41.6	8260
40	120	-360	20	504	-365	12488	-6447	-41.7	9468
50	160	-500	0	667	-508	16257	-9352	-41.1	12805
50	170	-360	20	557	-368	14185	-5986	-40.3	10085
0	10	80	-20	82	-92	1672	-2104	40.0	1888

TABLE C-XXVIII

LOAD CONDITIONS FOR STRAIN SURVEY, SPECIMEN NO. 14

Test Condition	Percentage Attained In Strain Survey	Moment at WS 220 IN. LBS. X 10^{-6}	Moment at WS 61 IN. LBS. X 10^{-6}	Shear at WS 220 Lbs.	C_p Of Shear % Chord	C_p Of Shear Fuselage Station
Test Condition 1	100%	10.00	14.77	30,012	30.10%	FS 540.22
Test Condition 2	100%	10.00	14.77	30,012	16.17%	FS 513.48
Upbending	50%	21.35	32.48	70,000	37.34%	FS 554.13
Downbending 1	50%	-17.57	-26.58	-56,666	40%	FS 559.24
Downbending 2	30%	-13.03	-27.12	-88,662	37.49%	FS 554.42

Rept. No. ER-11178 Acc. No.
Corp. Auth. Lockheed-Georgia Company

Title: C-130 - Results of Center Wing Residual Strength and Crack Propagation Test Program

Date: 15 Oct. 1971 PP: 295

Abstract: Fourteen C-130 center wings, each having from 4,000 and 13,000 flight of service use, were tested. Eight were cyclicly loaded to determine crack propagation characteristics and all 14 were loaded to destruction to determine their ultimate strength after service and simulated service use.

Location (G)
Class: Unclass.
Form GD80-5 CATALOG CARD ☐C ☐S ☐RD

Subjects: C-130
WINGS (AIRCRAFT)
CRACK-PROPAGATED
STRUCTURES
STRENGTH
STRESSES

Authors: F.L. Reeder/
W.J. Dirkin/
H.L. Snider

Cross Ref.: LSR No. 7221-70-012A
Contract No. NAS1-9485
Job No. 74C330

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Form GD80-5 CATALOG CARD ☐C ☐S ☐RD

Subjects: C-130
WINGS (AIRCRAFT)
CRACK-PROPAGATED
STRUCTURES
STRENGTH
STRESSES

Authors: F.L. Reeder/
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